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ABLATIVE HEAT SHIELDS FOR PLANETARY ENTRIES - A TECHNOLOGY REVIEW

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ABSTRACT: A review of ablation technology is carried out to assess the present state of the art and point out areas in which further research is required for planetary entry heat shields. Analyses and test techniques which have been developed to treat heat shields for Earth entry with combined radiative and convective heating are reviewed. With the lessons learned from Earth-entry research in mind, the work carried out to date for entry into various planetary atmospheres is reviewed and technological problem areas are discussed. In defining significant phenomena, various mechanisms and processes are discussed and their relative importance is illustrated by describing the analysis of a manned planetary-return Earth entry. In discussing the work to date on planetary entry, two broad categories of research are considered: (1) entry into tenuous atmospheres and (2) entry into the dense atmospheres of Venus and the giant planets. In each of these categories, atmospheric characteristics, entry velocities and modes, vehicle geometries, heating levels and candidate ablation materials are discussed. Present ground and flight-test capabilities are summarized and compared with planetary entry conditions. Particular attention is paid to coupled ablative-radiative phenomena which require the radiation spectrum of the facility to essentially duplicate that of the planetary atmosphere under study.

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In August of 1957, the Army Ballistic Missile Agency successfully reentered and recovered an ablating nose cone for a 1500-mile range IRBM, (1)* and the practicality of the ablative heat shield was demonstrated. In the succeeding 13 years, the ablative heat shield has evolved from a dramatically new concept to a proven, reliable and reasonably light-weight means of protecting entry vehicles from aerodynamic heating. During this time, ablation analyses have developed from simple steady-state solutions which consisted basically of transpiration-cooling theory with additional heat-absorption terms, to complex computer codes which describe the transient, coupled, in-depth response of a char-forming plastic to the heating from radiating, chemically reacting shock layers and boundary layers. This is not to say that all the significant problems in ablation analyses have been solved. We shall see that there are areas in which large uncertainties still exist and in which much work remains to be done. None the less, the past decade has produced great advances in ablation technology.

In the late 1950's, much research was carried out on subliming and glassy ablators. Most analyses were steady-state solutions since the bulk of the research was concerned with the ballistic entry of vehicles having fairly high ballistic coefficients (M/C_DA) and entry heat pulses characterized by high heating rates and short duration.

In the early 1960's the emphasis swung toward charring ablators, and lower density materials began to be developed for lower M/C_DA manned entry vehicles. Much work was done on transient, in-depth ablation analyses. Military vehicles were developed with higher and higher M/C_DA's. They penetrated deeply into the atmosphere before decelerating and hence their entries were characterized by extremely high stagnation-point heating, high pressures, transition to turbulent boundary layers, and high aerodynamic shears. Manned vehicles, on the other hand, tended to low M/C_DA's and lifting entries to keep deceleration loads within human tolerance levels. These vehicles decelerated high in the atmosphere and their entries were characterized by low pressures, low shears, laminar flow, modest heating rates, and long duration.

*The numbers in parentheses refer to the list of references appended to this paper.

Since the entry environments of these two classes of vehicles are so different, two separate and quite different areas of heat-shield technology have developed. In the case of high M/C_DA vehicles, high-density materials (such as high density nylon-phenolic, graphite-phenolic, carbon-carbon composites, and molded graphites) have been employed to resist mechanical erosion of the ablator surface, and to keep geometry change within reasonable bounds. In the case of manned vehicles, the trend has been to lower density materials since these materials provide superior insulating efficiency to protect against the long heat-soak characteristic of lifting, low M/C_DA entries. Because of the low shears and pressures associated with manned entries, it has been possible to develop ablators having low densities but which produce char layers with sufficient strength to resist mechanical failure within the flight envelope of the vehicle.

Because of its higher entry velocity, the Apollo vehicle encountered radiative as well as convective heating. In a sense, the Apollo project ushered in the era of radiative heating. Of course, research on shock-layer radiation preceded the actual Apollo entries by many years (work was actively under way in the late 1950's) but the Apollo project served as a focal point for much of the radiative heating research.

In the late 1950's, many types of facilities were used to test ablation materials. Oxyacetylene torches, combustion heated wind tunnels, pebble bed tunnels, rocket exhausts, plasma torches, and arc-heated wind tunnels were used with varying degrees of success. By 1962, the arc-heated wind tunnel had established itself as the most versatile and capable of the ablation test facilities.

Intensive research was carried out to produce improved arc heaters, capable of operating over wider ranges of pressure and enthalpy; and radiation sources were added to allow tests with combined convective and radiative heating. Today's ablation test facilities have reached a high level of refinement and have played an invaluable role in the development of ablation technology. In spite of their refinement, however, these facilities provide only a partial simulation of reentry flight conditions. Because of this, the proper interpretation of ground-test results in terms of in-flight ablative behavior remains one of the most important areas of ablation research.

Throughout the past 12 years, much research in polymer chemistry has been carried out in an effort to produce improved ablators. This work has enhanced our understanding of the pyrolysis process and char formation mechanisms, but no "super" materials have been produced. In general, advances in heat-shield performance have come more through understanding the significant phenomena of ablation than through development of greatly improved materials. When the effects of differing atmospheric compositions are accounted for, the phenomena involved in planetary entries and earth entries are the same. Hence, the technology developed for Earth entry heat shields can be used to, as a starting point, analyze the type of heat shields required for planetary entry vehicles.

In the present paper, the analyses and test techniques which have been developed to treat heat shields for Earth entry with combined radiative and convective heating are reviewed, and the significant phenomena are defined. The development of convective and radiative heat-transfer analyses, ablation layer analyses, and coupled heat transfer and ablation calculations are reviewed and, as an example of the current state of the art, a calculation technique developed for treating heat shields exposed to combined convective and radiative heating is described. With the lessons learned from Earth entry research in mind, the work carried out to date for entry into various planetary atmospheres is reviewed and technological problem areas are identified and discussed.

Symbols

A	projected cross-sectional area of entry vehicle
$\tilde{C}_A$	elemental mass fraction of ablation products
C_D	drag coefficient of entry vehicle
E_s	$4 \kappa_p \sigma T_s^4$
f	dimensionless stream function
g	acceleration due to gravity
H	density scale height
h	enthalpy
h_s	stagnation enthalpy
K_O	oxygen mass fraction
M	mass of entry vehicle
$\bar{M}$	molecular weight
$\dot{m}_g$	gas injection rate

P	pressure
P_{t_2}	stagnation pressure
Q_c	convective heat load
Q_R	radiative heat load
$\dot{q}_{BLK}$	reduction in heat flux due to convective blockage
$\dot{q}_c$	convective heating rate
$\dot{q}_{CHEM}$	heat flux due to boundary-layer chemical reactions
$\dot{q}_R$	radiative heating rate
R_{eff}	equivalent nose radius for convective heating
$R_{e,l}$	Reynolds number based on wetted length
r_b	cone base radius
r_N	nose radius
T	temperature
T_s	post shock temperature
V	velocity
z	altitude
Γ	radiation loss parameter $= E_s \, \delta \Big/ \frac{1}{2} \rho_\infty V_\infty^3$
γ_e	entry angle
Δh	enthalpy potential
δ	gas cap thickness
θ_c	cone half angle
κ_p	Planck mean absorption coefficient
ρ	density
σ	Stefan-Boltzmann constant
τ	aerodynamic shear

Subscripts:

e	entry condition
w	evaluated at the surface
∞	free stream

Convective and Radiative Heating for Earth Entry

Convective Heating

Convective heating has been a source of serious concern ever since the early days of supersonic flight. (Remember the press having a field day talking about aircraft crossing the sonic barrier and landing in the thermal thicket?) We will briefly review the development of nonablating heating calculations. The coupling of heating and ablation analyses will be discussed later.

Several specific areas can be pointed out in the literature on convective heating. In particular, the problems of stagnation-point heating, laminar heating, and turbulent heating calculations for very high-speed entry were put on a firm footing by the work of Fay and Riddell(2), Cohen(3), and Hoshizaki(4). More recently the work of De Rienzo and Pallone(5) permits extension of stagnation-point convective heating up to entry speeds of 21.33 km/sec. This last paper also gives an excellent summary of convective heat-transfer studies including the shock-tube experiments which formed the basis for much of the theoretical work. The work of Boison and Curtiss(6) and Zoby and Sullivan(7) explored the influence of body geometry on stagnation-point heating.

Laminar heat transfer on flat plates, sharp cones, and off the stagnation point on blunt bodies has been well established by the work of Cohen(3) and Lees(8). The results of Cohen have been correlated by Zoby(9) to put them in a more usable form.

Turbulent heating is still a subject for extensive research. The Reentry F spacecraft has provided an extensive amount of turbulent heating data which are still being compared with existing correlation equations. The earlier work of Zoby and Sullivan(10) showed that turbulent heating for a large number of flight vehicles, blunt and sharp, could be predicted with fair accuracy using reference enthalpy methods.

No discussion of turbulent heating is complete without a discussion of the problem of boundary-layer transition. Transition is known or believed to be influenced by Mach number, Reynolds number, leading-edge bluntness, surface roughness, and a host of other parameters including the ablation rate for an ablating surface. To date, no good correlation of boundary-layer transition is available. Any of the currently accepted correlations show a spread of the type shown in Figure 1 which is taken from Reference (11). Note that there is as much as an order of magnitude difference between the lowest and highest Reynolds numbers at which transition occurs. In addition, the influence of non-air gas mixtures on boundary-layer transition is not well known. However, barring major chemical effects which can significantly influence temperature and enthalpy profiles, there is no reason to believe that a transition correlation for air will not apply in non-air gas mixtures.

Radiative Heating

Radiative heating becomes an important design consideration when the gas-cap temperature exceeds approximately 10 000° K. A history of radiative heating calculations is given by Anderson(12) who points out that calculations using an optically thin or a gray gas model seriously overpredict the radiative flux to the wall. Figure 2, which is copied from Anderson's paper, shows how the state of the art for radiation calculations has changed in the past decade. Calculation procedures have progressed from the transparent, continuum only, uncoupled calculations used by many early investigators through the nongray, coupled but continuum only calculations of Hoshizaki and Wilson(13) to the coupled with continuum and lines calculations which are current today. In Figure 2, the last category is demonstrated by results from Page et al.(14) but could be equally well demonstrated with results from Wilson(15), Olstad(16), Chin(17), or Rigdon(18). The radiation model used in the calculation procedure described below is that of Wilson(19) which includes continuum and atomic lines. Anderson further points out that due to uncertainties in gas absorption coefficients, the radiative flux "can probably be determined within 50 percent" and if the uncertainty in the absorption characteristics of ablation products is also taken into account the radiative flux "is known within a factor of two." It is obvious that if the entry environment is dominated by radiative heat transfer the heat-shield weight requirement can be a serious uncertainty.

Most of the radiative heating calculations made to date and referenced above apply to the stagnation region. It was recognized prior to 1962 that radiative heating to conical bodies was negligible because of the cooling of the gas as it moved out of the stagnation region and onto the conical afterbody. However, bodies with larger cone angles have become intersting (e.g., Viking) and attention has again turned toward this problem. A very recent paper by Callis(20) shows results for sphere-cones with cone angles of 30°, 45°, and 60°. As anticipated, for the conditions investigated, the 30° and 45° cone heating rates fall to very small fractions of the stagnation-point values. The 60° cone, however, can experience radiative rates which may actually be higher than the stagnation-point values. Such results can have serious implications on both the heat-shield design and the aerodynamic characteristics of the entry vehicle. These results point out that radiative heating is a major unknown for the design of a vehicle flying in air. For vehicles flying in non-air gas mixtures, the uncertainty must be larger and requires further consideration.

Ablator Response

Ablation Analyses

During entry, an ablator responds to heat transfer and chemical attack from the adjoining shock layer. The ablator responds by receding, altering its internal temperature distribution, or both. The principal mechanisms of ablative response, as illustrated in Figure 3, are:

1. Sensible heat increase in the virgin plastic and char

2. Pyrolysis of the virgin plastic to form char and pyrolysis gases

3. Enthalpy increase in the pyrolysis gases as they flow through the char

4. Removal of the char surface by chemical attack and sublimation

5. Mechanical removal of the char

6. Reradiation of energy from the char surface

Most present-day ablation analyses assume one-dimensional flow of heat and mass normal to the local char surface. Such an assumption is valid for cases where the curvature of the heat shield is large compared to characteristic dimensions within the ablator (large nose radii), but for small ground-test models and for flight vehicles with small nose radii multidimensional solutions may be required. This is true in treating both temperature distributions and the internal flow of pyrolysis gases. At the present, complete multidimensional analyses have not reached a stage where they are suitable for routine use. Solutions have, however, been obtained for the important limiting cases of two-dimensional heat transfer with one-dimensional mass transfer (subliming ablators and graphite - no char layer)(21,22), and two-dimensional pyrolysis gas flow with one-dimensional heat transfer (models with rapidly varying external pressure distributions)(23). In the present paper, for the purpose of illustrating basic phenomena, we shall, for the most part, restrict our attention to the one-dimensional case.

A typical one-dimensional ablation analysis is composed of a finite-difference solution of the transient heat conduction equation throughout the char, pyrolysis zone, and virgin plastic; a set of empirical kinetic equations that describe the pyrolysis process; a chemistry routine to determine the state of the pyrolysis gases; a set of equations which describe char removal by chemical and mechanical means; and a char surface energy balance(24,25,26,27,28).

In the early days of ablation analysis, it was often necessary to sacrifice some rigor in the solution of the heat-conduction equation in order to keep computing times reasonably short. This problem has now been overcome by improved programing techniques and increases in computer speed so that today's finite difference solutions are essentially exact and require reasonably short machine times. At the present time, the accuracy of predicted internal temperature distributions is limited mainly by the material properties required as input data (i.e., specific heats, thermal conductivities, etc.). Virgin plastic properties can be measured with reasonable accuracy at temperatures where pyrolysis of the material does not occur. Since the techniques used to measure specific heats and thermal conductivities are steady-state procedures, measurement made on a material which is pyrolyzing must contain some uncertainties(29). Hence, material properties in the pyrolysis zone are difficult to obtain. The largest difficulties with input properties are associated with the char layer. Most ablation analyses use a char thermal conductivity which is a lumped parameter and represents the net conductivity of the char and the pyrolysis gases flowing through it. This lumped conductivity cannot actually be measured and is often inferred from arc tunnel tests of ablation models. That is, the char conductivity is adjusted until the predicted temperature distributions agree with those measured in the tunnel tests(30,31). In spite of the foregoing difficulties, present-day ablation analyses usually predict reasonably accurate internal temperature distributions. In reference 30 for instance, after adjustment of the char thermal conductivity, internal temperatures were predicted to within 30 percent of the measured values.

Pyrolysis Kinetics and Pyrolysis Gas Chemical State

The pyrolysis kinetics, as obtained from DTA (differential thermal analysis) and TGA (thermo-gravimetric analysis) data, generally predict pyrolysis rates and energy absorption with reasonable accuracy(32).

The state of the pyrolysis gases remains one of the major unsettled questions in ablation analysis. Its significance is illustrated in Figure 4. Early ablation analyses usually treated the pyrolysis gases as frozen at the composition at which they left the pyrolysis zone. If this assumption is made, then the pyrolysis gases will absorb only a small amount of energy in being heated from the pyrolysis temperature to the char surface temperature. Later, gas phase equilibrium computer codes were employed to treat the gases as being in complete chemical equilibrium (including the deposition of carbon in the hotter regions of the char). This equilibrium assumption results in the prediction of large heat absorption by the pyrolysis gases. Experimental data have been obtained which support both the equilibrium and frozen assumptions. Data obtained by Lundel(33) which agree with the frozen assumption are presented in Figure 4. A more accurate approach would be to carry out a kinetic analysis which employed measured rate constants for the important reactions. Such an analysis was reported in Reference (34) and indicated that the equilibrium assumption would significantly overpredict the heat absorption by the pyrolysis gases. The available rate data are, however, limited to relatively low temperatures and, in many cases, are available only for overall reactions rather than the individual kinetic steps involved in the pyrolysis gas reactions. Hence, a true kinetic model for the pyrolysis gas reaction has not been established and, lacking such a model, analyses based on low temperature data cannot be extrapolated to the high char temperatures associated with lunar return, planetary return, and planetary entry missions.

Char Surface Recession

Another important area of ablation analysis that is still largely unresolved is the prediction of char surface recession. Most analyses for manned vehicles employ the assumption that char recession is due entirely to chemical reactions and sublimation. This is usually justifiable within the flight envelope and makes the problem more tractable than if mechanical char removal had to be accounted for; but the question of char surface kinetics still presents formidable problems. Most analyses treat chemical char removal as being jointly rate and diffusion controlled. At low temperatures, where oxygen can diffuse to the char surface faster than it reacts, the recession rate is governed by the kinetics of the solid-gas reaction. At high temperatures, the rate of char recession is governed by the rate at which oxygen (and other reactive boundary-layer gases) diffuses to the char surface. Between these two regimes, there exists a transition region of joint rate and diffusion control. At yet higher temperatures, sublimation causes additional char recession. Most investigators have assumed that the rate controlled regime could be described by a single

Arrhenius equation with a constant activation energy. This surface recession model has received such wide acceptance that its validity is seldom questioned. However, data recently obtained for various graphites strongly suggest that the activation energy is, in fact, a decreasing function of temperature and that char surface removal may be chemically controlled to temperatures much higher than previously thought possible(35,36). Sublimation is another area of considerable uncertainty. Several investigators have shown that accurate solutions require the inclusion of high molecular-weight gas phase carbon species (C_1 through C_{10} usually). The thermodynamic properties of these large carbon species are not well established. Depending on their treatment of these species, various investigators have predicted significantly different sublimation rates at high temperatures and pressures(37,38). When, as in the case of high M/C_DA ballistic vehicles, mechanical char removal must be accounted for in addition to chemical removal, the prediction of char recession rates becomes uncertain indeed. Several theoretical treatments of char failure have appeared in the literature(39,40). Most of these analyses describe char removal as a cyclic process wherein the char builds up to a critical thickness and then is removed by a combination of thermal and internal pressure stresses. While this type of char failure has been reported for some materials, it is not encountered very often. The type of char failure that is most often seen in ablation tests involves the removal of relatively small fragments from the char surface. Figure 5 presents a typical set of test data for a carbon-phenolic ablation material. This type of char failure only occurs when the test stream contains oxygen. It is apparently a coupled chemical-mechanical process in which the char surface is weakened by oxidation and subsequently removed by aerodynamic shear. The phenomenon is particularly apparent under high-pressure test conditions and has been observed for graphites(41) as well as charring ablators(42). At the present time, this type of char removal must be described by empirical relations derived from ground tests.

Reradiation of Energy From Char Surface

The superiority of the charring ablator (over subliming and melting ablators) is largely due to its ability to reradiate large quantities of energy from a high-emissivity, high-temperature char surface. As will be shown in a subsequent section of this paper, this reradiation is one of the most important energy accommodation mechanisms for charring ablators. In most ablation analyses, the char surface is assumed to be a gray body and to have a constant emissivity. For most carbonaceous materials, this seems to be a reasonable assumption. However, materials which contain large amounts of silica often produce chars for which the gray-body assumption is significantly in error(43). The counterpart to emissivity is char surface absorptivity, which becomes important when the ablator is subjected to radiative heating. As with emissivity, the absorptivity is usually assumed constant, though its value is usually assumed to be different from that of the emissivity to account for the fact that the shock-layer radiation and the char surface reradiation involve different wavelength ranges. Actually, little is known about char absorptivity for short wavelength, high-energy (ultraviolet) radiation.

Since, reradiation from the char surface varies as the fourth power of surface temperature, it becomes more important as heating rates (and hence surface temperatures) increase. At a temperature of approximately 4000° K (for pressures of the order of 10 atmospheres), however, carbonaceous materials sublime. Hence, an upper limit on the amount of energy that can be reradiated exists (around 1200 W/cm^2). For entries which involve heating rates of a few thousand W/cm^2 (manned planetary return earth entry, for instance), char surface reradiation will usually be the most important energy accommodation mechanism. For entries that involve tens of thousands of W/cm^2 (which, as we shall see, may be the case for Jupiter entry), reradiation may be small compared to other ablative mechanisms.

Categories of Ablators

A great many different materials and formulations have been investigated for use as ablative heat shields. It might be expected that each of these materials would have its own special characteristics and would require a separate research program to determine its ablative behavior. To an extent, this is true, particularly with regard to thermophysical and thermochemical properties. In a broad sense, however, ablators may be meaningfully categorized according to their elemental chemical composition (i.e., the number of atoms of carbon, hydrogen, etc.). The basis of such a categorization is the concept of complete (gas-phase and gas-solid) equilibrium at the char surface. If such equilibrium is obtained, then, in a diffusion limited situation, the char recession rate (which is one of the major determinants of ablator performance) is completely determined by the vehicle geometry, the flight conditions, and the elemental composition of the virgin plastic. The assumption of complete char-surface equilibrium has been studied, both analytically and experimentally by many investigators(26,44,45,46). In many cases, it has been shown to provide an accurate prediction of ablator behavior. As mentioned previously, there are data available which suggest significant pyrolysis-gas nonequilibrium, and in such cases, the concept would not hold. As a guide to the overall categorization of ablators, however, the equilibrium concept is quite useful. When considered in this way, most ablators can be grouped into two classes: those which contain the elements carbon, hydrogen, nitrogen, and oxygen, and those which contain these four elements plus silicon. These two groups differ most significantly with respect to char chemistry. Materials which contain C, H, N, and O produce chars which are nearly 100 percent carbonaceous and whose chemical composition is virtually independent of test condition. Materials which contain Si, on the other hand, produce chars with surfaces composed primarily of solid SiO_2, liquid SiO_2, solid SiC, or solid C depending on the test condition. This behavior is particularly characteristic of elastomeric ablators and is illustrated by the results presented in Figure 6. Note that as the heating rate is increased, changes occur in both appearance and chemical composition of the char layers. Over this same range of test conditions, an ablator containing only C, H, N, and O would everywhere produce carbonaceous char. Silicon-dominated ablators (materials which behave as shown in Fig. 6) are best suited for use at low heating rates (low surface temperatures) where the char is primarily solid SiO_2 and, since it is an oxide, is virtually inert with respect to oxygen and hence is nonreceding. Carbon dominated ablators, on the other hand, are preferable at high heating rates since their carbonaceous chars do not suffer from the liquid layers and C/Si reactions characteristic of siliceous materials.

Coupling Between Heating and Ablator Response

One of the outstanding characteristics of an ablative heat shield is that it not only absorbs heat but, through the injection of gaseous ablation products, it also modifies the adjacent boundary layer and greatly reduces the level of aerodynamic heating. When the ablation products enter the boundary layer they are heated up, react with themselves, and (usually exothermically) with the boundary-layer and shock-layer gases; they thicken the boundary layer and alter its profiles of temperature, velocity, and species concentration, and they absorb radiation from the shock layer.

The exothermic reaction of the pyrolysis gases is both detrimental and beneficial. Because of its exothermic nature, it actually increases the heat flux to the ablator. However, these reactions also deplete the reactable oxygen in the boundary layer, reduce the oxygen flux to the char surface, and hence, in a diffusion limited situation, reduce the rate of char removal by chemical reaction. This combustive heating can be expressed as an effective increase in stagnation enthalpy. Hence, it is quite important at low flight speeds (low enthalpies) but becomes of lesser importance at high speeds. The oxygen depletion effect is important at both high and low speeds so long as chemical char removal is significant compared to removal by sublimation and mechanical processes. In treating the ablation product boundary-layer reactions there is, once again, the question of reaction rates. The limiting cases of equilibrium and frozen chemistry are, of course, available. At higher boundary-layer and shock-layer temperatures (say above 2000° K), equilibrium is usually assumed since reaction rates are expected to be fast. A considerable amount of data supporting this assumption has been published, but in many cases the frozen assumption seems to give better results. The analysis that was most successful in predicting the performance of the Apollo heat shield treated the ablation products in the boundary layer as being frozen.

The thickening of the boundary layer and the alteration of its profiles constitutes the well-known "transpiration cooling effect." This is an important ablation mechanism under virtually all flight conditions, and becomes particularly important under conditions producing high radiative heating rates. Under these conditions, the resulting high mass injection rates "blow the boundary layer off" the char surface and reduce the convective heating to zero. This phenomenon is usually illustrated by a plot of the ratio of convective heat flux (or Stanton numbers) with and without blowing versus a nondimensional mass injection rate. From such a plot, presented in Figure 7, it is seen that the convective heat flux tends to zero with increasing blowing rate as shown by the solid curve. Experimental ablation data have been presented, however, indicating that, with increasing blowing rate, the convective heating approaches a finite asymptote (0.2 to 0.3 of the no-blowing value) rather than zero[47]. This type of behavior should be expected when the blowing is produced by an ablator responding to purely convective heating. If the heating were actually reduced to zero, the ablator would stop ablating, the blowing rate would go to zero, and, with no mass injection, the heating rate would increase to its no-blowing value. Hence, some nonzero asymptote must be approached in convective heating situations. This type of behavior will not occur, however, in the presence of large radiative heating rates. In this case, the high mass injection rates (ablation rates) resulting from the radiative heating can, indeed, drive the convective heating rates to extremely low values.

Until a few years ago it was generally assumed that the gaseous ablation products would not absorb a significant part of the radiation from the hot shock layer. Then, a number of investigators carried out analyses in which the absorption by the ablation products could be accounted for[48,49]. They found that the ablation products could actually be rather efficient absorbers (especially at ultraviolet wavelengths) and predictions of up to 50 percent reduction in radiative heat flux were published. Recently analyses of this type have been further refined and now reductions of around 25 percent are being predicted for typical planetary return

Earth entries(50). Analyses of this type are discussed in the following section of this paper.

State-of-the-Art Analysis for Planetary Return Earth Entry

In order to identify those ablative mechanisms which are most important during a high-speed entry, a typical up-to-date analysis is described and typical results for planetary return Earth entry are reviewed.

A state-of-the-art analysis which is being used to study some of the problems associated with Earth entry of a blunt body at planetary return speeds is given by Smith et al. in Reference (50). Figure 8 is taken from that reference to show the logic involved in developing such an analysis.

The analysis must begin with a trajectory if a transient ablation solution is required. If only simple, point calculations are required then velocity and altitude or velocity and density are all that are necessary to initiate the analysis.

The inviscid radiating solution is used to calculate all the flow parameters in the subsonic portion of the flow field behind the bow shock. In the analysis shown in Figure 8, the method used is that of Suttles(51) who combined the one strip integral method(52) with the radiation model of Wilson(19). The output from the inviscid flow-field program is used to provide local values of pressure, temperature, density, and velocity. This information can be used as local edge conditions with a conventional boundary-layer solution to provide initial estimates of the convective heating. In the analysis of Reference (50), these convective heating calculations were made using the correlation equations of Zoby(9) for equilibrium air.

The radiative flux calculated by the inviscid flow-field program and the convective heating rate calculated by the boundary-layer program are used as inputs to the ablation program (developed by Kendall, Rindal, and Bartlett(26)) which computes the transient, one-dimensional response of a charring material with heat conduction and in-depth pyrolysis governed by Arrhenius type rate equations. The pyrolysis gases are assumed to be in chemical equilibrium throughout the char, and two-phase chemical equilibrium is assumed at the char surface. All char recession is assumed to be caused by rate and diffusion-limited chemical reaction and sublimation.

When the ablation rates have been calculated, it is possible to compute the velocity profiles in the boundary layers using the technique described in Reference (50). By testing the resulting stream function against a limiting value, the analysis decides whether or not the ablation rate is large enough to move the air-ablation products boundary layer away from the wall and replace it with an ablation products layer with an almost constant, low-value shear at the wall. If the ablation rate is sufficiently large the strong injection solution is used. This solution, described in detail in Reference (50), assumes that the boundary layer consists of a layer of ablation products next to the wall and a layer within which the viscous effects adjust the flow variables from the inner layer values to the inviscid outer layer values. In the layer next to the body, constant shear and negligible conduction are assumed. In the viscous layer, the elemental composition is computed by assuming a cubic variation from the ablation products composition in the inner layer to the air composition at the edge of the inviscid outer layer. If the ablation rate is moderate, a conventional boundary-layer solution is used in which the stream function at the wall is defined by the ablation rate.

The inviscid outer layer and the boundary layer for either strong or moderate injection are coupled to provide continuous enthalpy profiles across the shock layer. First, the inviscid layer is displaced from the wall by a distance equal to the displacement thickness of a boundary layer with mass addition. The boundary-layer edge enthalpy value is matched to a value in the inviscid layer near the edge of the boundary layer. This information provides a smooth enthalpy profile which is used by the radiation model to recompute the radiative flux. This flux is used as input for the next iteration on the mass loss rate calculations. The solution continues until a consistent set of flux and mass loss rate values is obtained.

There are several radiation models which might be used for calculation of radiative transfer through the ablation products layer. As previously noted, the radiation model used in Reference (50) was that of Reference (19). This model contains the major features for air radiation as well as the prominent features from a large number of the chemical compounds resulting from an equilibrium analysis of the ablation products from common ablators such as phenolic nylon and phenolic carbon. Thus, when this code is coupled with the boundary-layer solutions described above, the major influence of the ablation products on the radiative heating has probably been accounted for. The answers should certainly be within the factor of 2 mentioned by Anderson. Typical results from analyses of this type are presented in Figures 9 and 10. Figure 9 shows the extent of the ablation products layer and the temperature distribution through the shock layer, and presents the calculated shock stand-off distance, heating rates, and the char-surface temperature for conditions typical of peak heating during a manned planetary return Earth entry. Blockage by ablation products reduces the inviscid heating by about 22 percent. It is of interest to see in which spectral regions the absorption is taking place. This is shown by the difference between the dashed and solid lines in Figure 10. Note that the absorption is almost entirely in the ultraviolet region.

Only by carrying out a coupled analysis such as that just described is it possible to accurately assess the heat-shielding requirements for severe radiation-dominated entries. Coleman, et al., carried out a thorough study of heat-shielding requirements for blunt and conical manned planetary return entry vehicles. Their analysis included all the important mechanisms mentioned previously. For a baseline theoretical model (which assumed a phenolic-nylon ablator, no mechanical char failure, equilibrium pyrolysis gases, laminar flow, equilibrium shock-layer chemistry, and no blockage of radiation by ablation products), they showed the magnitudes of the various heat absorption and blockage mechanisms as a function of time during entry. These data are reproduced (with some modification) in Figure 11. In constructing Figure 11, the data of Coleman et al. (49) were modified to reflect a 25-percent radiation blockage due to ablation products since, in light of recent developments, this is believed to be realistic. Results are presented for the center line of an Apollo-like vehicle. For the blunt Apollo-like vehicle, radiation is the dominant heating mechanism, and the significant energy accommodation mechanisms are radiation blockage by ablation products, transpiration cooling, surface reradiation, and the enthalpy increase in the pyrolysis gases. For conical vehicles, the dominant heating was found to be convective, and the significant energy accommodation mechanisms were transpiration cooling, reradiation, and pyrolysis enthalpy increase.

Ablative Response in Turbulent Flow

All the results presented thus far are rigorously applicable only to laminar flows. Actually, little is known about the response of ablators subjected to turbulent flows. Since arc tunnels are generally low-density facilities and can accommodate only small models, ordinary ablation tests never produce Reynolds numbers high enough to produce turbulent flow. Some ablation tests have also been carried out in turbulent pipe-flow and channel-flow facilities and have produced valuable data, but because of radiation exchange with facility walls and difficulties in precisely defining the flow, the application of these results to flight conditions is not straightforward. Some turbulent ablation data have also been obtained from flight tests. Data for several materials having densities from about 0.3 to 1 gm/cm^3 were obtained from the NASA Pacemaker series of small flight vehicles and data for a variety of high-density materials have been obtained from flights of military vehicles. Unfortunately, Mach numbers and enthalpies over which these data were obtained are considerably lower than those for severe planetary entries.

The analysis of the response of an ablator to turbulent flow is usually done by calculating the convective heating rates by available turbulent boundary-layer techniques and reducing the convective blocking effectiveness of the ablation products to 1/3 or 1/4 of their laminar values. Very little in the way of coupled radiative-convective analyses with turbulent flow has been attempted thus far since it has generally been assumed that radiative heating will be insignificant in the afterbody regions where the flow could be turbulent. In general, the approach used with nonmilitary vehicles has been to choose vehicle geometries and entry trajectories so as to avoid turbulent heating. This same approach will probably be used with planetary entry vehicles.

Planetary Entries

Ablation research for Earth entry has been reviewed. Now we shall use this background and consider the problems associated with planetary entries. From our review of Earth entry research, we have seen that ablative heat shields provide reliable, reasonably lightweight thermal protection systems, but because of uncertainties in several key technology areas (boundary-layer transition, chemical state of pyrolysis gases, char-surface recession mechanisms) a conservative design approach must be used. At the present time, these same uncertainties will require conservative designs for planetary vehicles. For severe Earth entries, the most significant ablative energy accommodation mechanisms were seen to be blockage of convective and radiative heat flux by ablation products, char surface reradiation, and pyrolysis enthalpy increase. Various planetary entries will now be considered and the most significant ablation mechanisms identified in each case.

At this point it should be pointed out that prelaunch and transit environments such as sterilization, vacuum exposure, and cold soak can significantly influence the choice of an ablation material for a particular mission. The study of these effects constitutes a broad technology area in itself and is outside the scope of the present paper. It must be remembered, however, that the material properties used in the analyses described herein must be those that the ablator possesses after its interplanetary trip, just prior to atmospheric entry.

In Table 1, properties of various planetary atmospheres(53,54,55,56,57) and unpublished Project Viking atmospheric models are presented*. The near planets may be grouped into two categories, those with atmospheres considerably less dense than the Earth's and atmospheres more dense than that of Earth. In the low-density group, the planets of most current interest are Mars and Mercury. In the high density group, they are Venus and Jupiter.

In the present paper, we shall consider Mars as being representative of the planets with tenuous atmospheres. Pritchard(58) has considered the possibility of using a vehicle originally designed for Martian entry to explore Mercury, Titan (a moon of Saturn), and other planets with thin atmospheres. He concluded that, for surface pressures as low as 1 mb, an entry vehicle designed for Mars could be used for Mercury and Titan missions with appropriate changes in the terminal descent system and, perhaps, in the guidance system. Hence, Mars is a good focal point for low-density-atmosphere entries.

In the case of entry into dense atmospheres we shall consider two planets, Venus and Jupiter. Venus is selected for consideration since it is unique in posing entry problems somewhat comparable to those of Earth entry. Jupiter is chosen as being characteristic of the giant planets (Jupiter, Saturn, Uranus, Neptune). This is done for two reasons. First of all, interest in Jovian entry is currently high and, as a result, much more information is available for Jupiter than for the other giant planets. Second, Jupiter is thought to possess many of the atmospheric characteristics that would be encountered during Saturn, Uranus, and Neptune entries.

Entry Into Tenuous Atmospheres - Mars

The essential problem associated with entries into tenuous atmospheres is that of decelerating to conditions at which parachutes or other high drag devices can be deployed. This must be accomplished at altitudes which are high enough to allow atmospheric sampling in most planetary exploration missions. In order to achieve this high-altitude deceleration, low values of the vehicle ballistic coefficient, M/C_DA, are required. As pointed out by Roberts(59), deceleration within a given atmosphere requires that the atmospheric drag parameter,

$$P/(Mg/C_DA)$$

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* The values presented in Table 1 are intended only to roughly categorize the various atmospheres. It is recognized that many of these values are uncertain and debatable.

be less than approximately 10. Hence, a vehicle with an M/C_DA of 314 g/cm^2 will be decelerated by the Earth's atmosphere whereas, deceleration in the Martian atmosphere will require M/C_DA's on the order of 3.14 g/cm^2.

Atmospheric Characteristics

Our concept of the Martian atmosphere has changed drastically in the past few years. When Martian entry vehicles began to be seriously studied in 1961-62, it was recognized that much uncertainty existed regarding the pressure and composition of the Mars atmosphere. Early estimates of surface pressure ranged from 10 to 30 mb (0.01 to 0.03 atm), and compositions from 80 percent nitrogen, 20 percent CO_2 to 100 percent CO_2 were considered possible. In that time period most studies assumed significant amounts of both N_2 and CO_2. Recent spectroscopic data and measurements of the occultation of radio-frequency transmission from the Mariner flyby spacecraft have, however, considerably improved our knowledge of the Martian atmosphere. The Project Viking Mars Engineering Model Working Group has reviewed these data and proposed a set of model atmospheres with surface pressures ranging from 4 to 10 mb (0.004 to 0.01 atm), compositions from 100 percent CO_2 to 71 percent CO_2, 29 percent Ar, and density scale heights (in the stratosphere) from 4.5 to 12 km. These various atmospheres correspond to maximum and minimum surface densities and a mean model. Density and temperature profiles for these model atmospheres are presented in Figures 12 and 13.

Entry Vehicle Geometries, Entry Modes, and Velocities

By far the most often used geometry for Martian entry vehicles is the spherically blunted cone. This configuration has the advantages of low ballistic coefficient, good payload packaging characteristics, and good aerodynamic stability. The Project Viking entry capsule is shown in Figure 14 as an example of this type of configuration.

Nearly all the studies carried out to date have considered ballistic entries. While many early investigations featured vertical or near-vertical entries, the trend in recent years has been toward entry angles of -20° or less. For low ballistic-coefficient vehicles ($M/C_DA = 3 \rightarrow 5$ gm/cm^2) entering at shallow angles, heating rates tend to be very low. The maximum heating rates on such vehicles as calculated in References (60), (61), (62), and (63) are summarized in Table 2. From this table it is seen that maximum heating rates range from 100 to 200 W/cm^2 for direct entry and are on the order of 20 W/cm^2 for entry from orbit. Furthermore, radiative heating is inconsequential for entries less than about 7 km/sec. Hence, for the Martian entries of greatest current interest (e.g., orbital entries such as Project Viking), the heat-shielding problem is not severe. Only in the case of very high-speed advanced missions (such as the 11.5 km/sec Mars Surface Sample Return mission mentioned in Reference (62)) do the heating rates become high enough to really exercise present-day ablators. Since high-speed entries of this type have much in common with Venusian entries, they will not be discussed in this section.

Since the heating rates are so low, the selection of an ablative material for a Martian entry vehicle is influenced more by its ability to withstand such environments as prelaunch sterilization, in-transit cold soak, solar radiation, micro-meteorite bombardment, and vacuum exposure(63,64) than by its high ablative efficiency. From an ablative performance standpoint, the outstanding requirement is that the material be of low density and possess good insulating capability. Filled elastomeric materials seem to be well suited to these requirements. A typical material might be composed of silicone resin, silica microspheres, phenolic microballoons, and carbon and silica fibers. This type of material has been studied and tested extensively(46,65,66,67) and, for low heating-rate applications, is well proven.

The heat shield remains an important part of the overall vehicle design for a Martian entry vehicle. Most design studies(67,68) indicate that the heat shield will comprise 10 to 20 percent of the entry vehicle gross weight. The materials are developed and available, however, and the overall problem seems well within the present state of the art.

Entry Into Dense Atmospheres

As shown in Table 1, the near planets may be grouped according to atmospheric density. The planets of immediate interest which have atmospheres as dense or more dense than that of Earth are Venus and the giant planets, Jupiter, Saturn, Uranus, and Neptune. In most prior research, Venus and Mars have been considered together because of their similar atmospheric composition. If comparable entry velocities are considered for the two planets, this would be a logical grouping. In most cases, however (because of the difference in mass of the two planets), Venusian entry velocities are much higher than those for Mars. As a result, Venusian entries usually entail severe radiative and convective heating and pose a significant challenge to present-day ablative heat shields. As discussed in the preceding section, this is usually not the case for Mars. Except for the fact that they both pose appreciable heat-shielding problems, however, Venus entries and entries into the atmospheres of the giant planets have little in common. Hence, they will be discussed as separate categories.

Venus

Atmospheric Characteristics

Data obtained from the flights of Mariner V and the Russian Venera 4 vehicles have greatly reduced the uncertainties associated with the density, temperatures, and composition of the Venusian atmosphere. From analyses of these data(55), it appears that the altitude from which Venera 4 sent its last radio transmission is in doubt. Originally it was reported that Venera 4 had measured pressures of from 16.4 to 20.3 atmospheres at the Venusian surface. However, if it is assumed that the last Venera 4 transmission came from an altitude of approximately 30 km, then the Venera 4 data fit nicely with the Mariner V data, define the temperature and pressure profiles in the Venus atmosphere quite well, and surface pressures of 167 atmospheres are indicated. This interpretation of the data is appealing but there are still arguments pro and con. As a result, model atmospheres have been developed corresponding to surface pressures of 16.4 and 167 atmospheres and various degrees of solar activity. Density and temperature profiles for these atmospheres are presented in Figures 15 and 16. The composition of the Venusian atmosphere is now believed to be between 90 percent CO_2, 10 percent N_2, and 100 percent CO_2. From an entry heating standpoint, the atmosphere is reasonably well known since the important parameters are composition and scale height, and the scale heights in the sensible (from a heating standpoint) atmosphere are nearly the same for the various models shown in Figure 15.

Entry Velocities, Modes, and Vehicle Geometries

The entry velocities encountered for various missions range from 11 to 15 km/sec. The lowest velocities correspond to entry from orbit, and the highest velocities are those for a Mercury swing-by mission. For direct entries, as shown by Norman and Hart(69), the velocities can range from 11.6 to 13 km/sec depending on the year of the mission. Some early studies considered vertical entries but most recent investigations have studied entry angles from -30° to -50°. The Venus missions studied so far have utilized ballistic entries since it appears that heating rates can be kept within a manageable range without the use of lift and there is no problem in decelerating to subsonic speeds at sufficiently high altitudes for atmospheric studies.

The geometries considered by almost all investigators are, as was the case for Martian entry, spherically blunted cones. In general, nose radii and cone angles for Venusian vehicles are smaller than those used for Mars entry. This trend toward sharper, more highly sweptback vehicles reflects attempts at minimizing the total heat load by reducing radiative heating. As shown by Norman and Hart(69), however, not all the effects of reducing nose radii are favorable. If transition to turbulent flow occurs (in Ref. (69), the assumption of a momentum thickness transition Reynolds number of 250 results in turbulent flow at the cone edge just prior to peak heating), then the thick entropy layer caused by a large nose radius can significantly reduce the levels of turbulent heating and shear. Most studies have indicated nose radii on the order of 1 foot and cone angles from 50 to 60° (Fig. 14).

Heating Levels

Studies of stagnation point and laminar convective heating in various atmospheres(70,71,72,73,74) have shown that CO_2 and CO_2/N_2 mixtures will produce nearly the same (actually about 10 percent higher) heat fluxes as air for comparable vehicles and flight conditions. It is to be expected that turbulent heating levels will be comparable for these gas mixtures.

On the other hand, radiative heating from air and from CO_2/N_2 mixtures differs considerably(75,76,77,78). One reason for this is that the temperatures at which these gas mixtures begin to radiate strongly are quite different. For example, at temperatures of approximately 8000° K, where air radiates weakly, CO_2 radiates strongly. Another reason is that because of the different radiating species involved the spectral distributions of radiation from these gases are quite dissimilar. This is shown in Figure 17 which was prepared using the radiation model of Nicolet(79). The reference conditions for this figure were selected from Reference (80) with a V-3 atmosphere(55) and the calculations were carried out for an isothermal slab. Three gas mixtures are shown: air, 90 percent CO_2 with 10 percent N_2, and 100 percent CO_2. The calculated equilibrium post-shock temperature varies between 9090° K for air and 9540° K for 100 percent CO_2. Note that there is almost no difference between the CO_2 and CO_2/N_2 spectral distributions except in the region from 3.0 to 4.5 eV. Nitrogen and cyanogen bands dominate in this region so this difference represents the effect to be expected by the presence of nitrogen in the atmosphere. This difference, while not negligible, will not make or break a design and, conversely, designs which allow for a reasonable nitrogen content will not be unduly penalized. It is evident that CN, while an important radiator, is not the dominant radiator which it was originally believed to be(75,81) for earlier Venus atmospheres which contained as much as 60 percent N_2.

A comparison of both of these atmospheres with the air results on Figure 17 shows significant differences. First of all, it is noted that the air temperature is significantly lower than either of the others so that the radiation is expected to be lower. However, the difference shown indicates the importance of CO(4+) which dominates the spectral region from approximately 5 to 11 eV. In air there are no comparable radiators in this region and, hence, the spectral distribution and radiative flux encountered during a comparable Earth entry will be significantly different from any Venusian entry.

Figure 17 suggests the importance of using an ablator which will absorb the CO(4+) radiation. As shown in Figure 10 for air, the absorption by CO in ablation products is very noticeable in the spectral region from 5 to 11 eV. However, for air there are no prominent radiation sources in this frequency band and the effectiveness of the ablation products is minimized. These results suggest, however, that any ablator chosen for a Venus entry should be tailored to provide large quantities of CO in the ablation products so as to minimize the radiative heat load.

In Table 3, heating rates calculated by various investigators(60,69,75,80) for various Venus missions are presented. From this table, it can be seen that the maximum heating rates for out-of-orbit and direct entries are in the same range as those for high velocity Earth entry missions. For the out-of-orbit and direct entries, it is also seen that the pressures and aerodynamic shears are higher than those characteristic of manned entries but much less than those typical of high M/C_DA ballistic Earth entry vehicles. The duration of heating for these Venus entries is relatively short (on the order of 30 sec).

Candidate Materials

The levels of heating rate, pressure, and shear presented in Table 3 can be used to select a class of ablators for Venus entry missions. First of all, the high heating rates shown in Table 3 prohibit the use of silica-dominated materials such as those used for Martian entry vehicles. At high surface temperatures, chars which contain large amounts of SiO_2 will recede rapidly due to SiO_2 melting, Si/C reactions, or both. At these temperatures, char surface reradiation will be a primary energy accommodation mechanism and so a high emissivity, stable carbonaceous char layer is required. All of this dictates the use of a carbon-dominated material. As mentioned previously, the pressures and shears associated with Venusian entry are somewhat higher than those typical of manned entries. Because of this, mechanical char failure may become a significant consideration, at least for selected regions of the entry vehicle. In Earth entry research the usual approach has been to increase ablator density in an attempt to obtain increased char strength. To a degree this has been successful, but large increases in ablator density have been found to result in only small changes in the test conditions required to produce mechanical char removal. As mentioned previously, the most commonly encountered type of mechanical char failure involves a weakening of the char skeleton by chemical reaction followed by removal of discrete char fragments by aerodynamic forces. This is not completely understood and is not analytically describable at the present time. It is probable that an ablator of somewhat higher density than those used on manned vehicles should be used on a Venusian entry vehicle. A density of about 0.7 gm/cm^3 seems a reasonable choice. The use of high-density materials is not desirable since their resistance to mechanical char failure is only slightly greater than that of the lower density materials, and their inferior insulating efficiency (resulting from their high density) would require large increases in heat-shield weight.

Hence, a carbon-dominated material of medium density is indicated. Some of the best-known materials of this type may, however, be unsuitable for planetary missions. Most phenolic-based and many epoxy-based materials have been found to be incapable of withstanding the prelaunch and transit environments of sterilization, vacuum, and cold-soak which were so influential in the choice of an elastomeric material for Martian entries. A possible approach may be to use a material composed of an elastomeric resin, carbon microspheres, and carbon fibers in an effort to utilize the good low-temperature properties of the resin; and with the high carbon-filler content, produce a stable high-temperature char. It would be interesting to know what type of ablator the Russians used on Venera 4.

Present State of the Technology

Since the levels of heating, pressure, and shear during Venusian entry are significantly higher than those for Martian entry, the heat-shield analysis and design is a more critical part of the overall vehicle design. Except for the most severe entries (Mercury swing-by), however, the entry conditions are close enough to those of high-speed Earth entries that the technology already developed for Earth entry is largely applicable. The situation then is that while there are significant uncertainties in both heating and ablation analyses, it appears that, through conservative design, workable heat shields can be produced within the present state of the art. Present indications are that for out-of-orbit and direct entries the thermal protection system will constitute from 10 to 25 percent of the entry vehicle weight. (A recent study by Jaworski and Nagler(82) indicates that, for the less severe Venusian entries, the weight of the actual ablation material may be less than 5 percent of the total vehicle weight. These numbers seem small and bear further study.)

In the analyses of aerodynamic heating, there are significant uncertainties associated with the prediction of boundary-layer transition, turbulent heating, and radiative heating levels. In predicting ablator response, the major uncertainties appear to be related to the kinetics of the pyrolysis gas reactions, ablator response to turbulent flows, the prediction of char recession rates, particularly recession due to mechanical char removal.

The prediction of boundary-layer transition has, for many years, been recognized as one of the most important problems in aerospace research. It is also one of the most difficult. In view of the great amount of research already devoted to the problem, it seems unrealistic to expect any breakthroughs in this area. Research should, of course, be continued, but in designing near-future planetary entry vehicles, we may have to accept transition Reynolds numbers with order-of-magnitude accuracies. On the other hand, it appears that the present uncertainties in radiative heating levels could be reduced by carrying out completely coupled analyses, such as those described earlier for high-speed Earth entry, in which all the significant phenomena such as self-absorption, radiation cooling, realistic (lines, band systems, and continuum) radiation models, and radiation blockage by pyrolysis gases are accounted for. In practically every study carried out so far for Venusian entry, one or more of these phenomena have been neglected. With regard to the uncertainties in ablator response, the greatest need is for more and better experimental data to better define the important mechanisms and to lead to more physically realistic analyses.

Jupiter

Atmospheric Characteristics

Our knowledge of the atmosphere of Jupiter is much less complete than for Mars or Venus. Because of the Jovian cloud layers, the planet's surface (if one exists in the usual sense) cannot be observed. The available data concern the atmosphere above the cloud layer. The makeup of the Jovian lower atmosphere is largely a matter of speculation. In 1967, Michaux, et al. (57), reviewed the status of our knowledge of Jupiter. They presented a possible makeup for the lower atmosphere and Jovian interior (based on models by Gallet and Peebles) which is presented in Figure 18 of the present paper. From this figure it is seen that the atmosphere is pictured as becoming denser and denser until a surface which may be liquid or solid is reached. From the standpoint of reentry vehicles, however, it is the atmosphere above the clouds that is of most interest since vehicles with moderate M/C_DA's will decelerate to terminal conditions before reaching the clouds.

Table 4 presents a sampling of proposed Jovian model atmospheres(56,83). As shown in Table 4, there is general agreement that the main atmospheric constituents (above the cloud layers) are H_2 and He, but the proposed compositions range from predominantly H_2 to predominantly He. Proposed cloud-top pressures and temperatures range from 2 to 24 atmospheres and 150 to 168° K; the atmospheric scale heights range from 12 to 21 km.

Entry Velocities, Modes, and Vehicle Geometries

For Jupiter, escape velocity is approximately 60 km/sec and surface satellite speed is approximately 40 km/sec. Accordingly, most studies of Jovian entry have considered this range of entry velocities. The equatorial velocity of the planet is, however, about 13 km/sec and, hence, entry in the direction of planetary rotation and in the vicinity of the equatorial plane could reduce the relative direct-entry velocities to about 50 km/sec.

Most studies to date have considered ballistic entries and, again, the most widely considered configuration is the sphere cone. Because of the very high speeds and heating rates involved in Jupiter entry, there are some indications that guided (lifting) entries may be required to keep the heating within manageable limits(83). Tauber(83) has studied the relative merits of blunt and conical vehicles for Jovian entry. His conclusion is that the conical vehicles have a clear advantage with regard to required heat-shield weights. In fact, he suggests that if the atmosphere were primarily H_2, a blunt (Apollo-like) vehicle might have to be composed almost entirely of heat shield. Present indications are that the cone angles for Jovian vehicles will be smaller than those for Martian and Venusian vehicles (Fig. 14). The most advantageous cone angle and, for that matter, the overall character of the entry heating appear to depend strongly on the atmospheric composition. This dependence will be discussed subsequently.

Heating Levels

Heating rates, for most Jovian entries, are predicted to be extremely high by Earth-entry standards. Tauber(83) studied the effects of vehicle geometry and atmospheric composition on radiative and convective heating rates for a spherically tipped, conical vehicle. Some of his results are presented in Figure 19. The results presented in Figure 19 were computed for a vehicle having a 30° cone half-angle, a base radius of 1 m, and a nose radius of 1 cm entering on a shallow ballistic trajectory at 50 km/sec. During entry, boundary-layer Reynolds numbers were limited to 5×10^6 in order to insure laminar flow. Both radiative and convective heating rates are presented for the stagnation point and the conical afterbody, as a function of atmospheric H_2 content. As shown by Figure 19, a 100-percent H_2 atmosphere would produce heating rates that are sizable but comparable to those experienced during severe Earth entries. A He atmosphere, on the other hand, would produce radiative and convective heating rates far beyond those with which we have had experience. At the present time, it is difficult to pick a "most likely" Jovian atmosphere. The most recent papers on the subject do, however, seem to be indicating higher and higher H_2 contents. In 1963, Spinrad and Trafton(84) proposed a 60-percent H_2 atmosphere and, in 1967, Beckman(85) proposed an 82-percent H_2 atmosphere. If these high H_2 contents prove to be correct, the Jovian entry heating problem will be large but not unmanageable.

The theory of laminar convective heat transfer for H_2/He mixtures appears to be on fairly firm ground. Stagnation-point heating in H_2 has been treated by Scala(86) and Marvin and Deiwert(71), and correlated by Zoby(70). Stagnation-point heating in He was measured and correlated by Pope(87). Zoby(70) has proposed an approximate method (involving summation of terms weighted according to the mass fractions of the individual gases) for calculating stagnation-point heating in gas mixtures, and recent unpublished work by Sutton and Graves at the Langley Research Center has shown that the summation approximation gives accurate estimates of heating for Jovian atmospheres. With regard to laminar heating away from the stagnation point, Marvin and Deiwert(71) found that heating-rate distributions were only affected to a minor degree by gas composition. The prediction of transition and turbulent heating in H_2/He mixtures involves large uncertainties, just as it does in the case of air.

The radiation from H_2/He shock layers is significantly different from that produced by air. This is illustrated by Figure 20. In Figure 20, spectral flux is presented as a function of photon energy for a 61-percent H_2, 36-percent He shock layer, and for an air shock layer. The results shown for the Jovian atmosphere were taken from the work of Stickford and Menard(88). They were obtained by carrying out an adiabatic calculation, which included self-absorption, for a 12 000° K, 1-cm-thick layer. The results shown for air are those previously presented in Figure 10(a) and include self-absorption, radiation loss, and, as indicated by the short dashed curves, absorption by ablation products. The H_2/He results include Lyman and Balmer line radiation while the air results are for continuum only. However, as Figure 10(b) indicates, the line radiation from the air will not change the comparison shown in Figure 20. The temperature of the air shock layer (12 900° K) is comparable to that of the H_2/He layer, but the air layer is nearly an order of magnitude thicker. From these curves it is seen that, for comparable shock-layer temperatures, the intensity of radiation will be much greater for the H_2/He mixture than for air. Also, it can be seen that the relative amount of radiation in spectral regions not subject to absorption by ablation products is greater for H_2/He than for air. This suggests that absorption by ablation products may not be as important for Jovian entry as it is for Earth or Venus entry. Wilson(15) showed only about a 10-percent reduction in radiative heating due to ablation products.*

Candidate Materials

The extremely high heating rates and relatively high pressures that are predicted for Jovian entry (see Fig. 19) immediately suggest the use of high-density, carbonaceous materials. Graphites, carbon-carbon composites, and high-density graphite fiber reinforced plastics have all been suggested for use on Jovian entry vehicles. In the nose regions of the vehicle, such materials will probably be used. The requirement of reasonably small shape change, alone, may dictate high-density materials in this region. On the vehicle afterbody, moderate density ablators appear feasible. Materials similar to those discussed for Venusian entry may work well for these applications.

Actually, there are several basic phenomena which must be investigated before any reasonable choice of materials can be made. The first of these concerns the response of materials to very high radiative heating rates. At present, no realistic tests have been carried out at heating rates over a few kW/cm^2. It has been suggested that materials may be torn apart by the high thermal stresses and internal gas pressures that heating rates on the order of tens of thousands of W/cm^2 will produce. Such a possibility cannot be discounted. What is needed is test data at these conditions. Second, there are questions concerning the absorptivity of ablative char layers in the different spectral ranges. Wilson and Spitzer(43) measured the absorptance of several chars and graphites at temperatures up to 3400° K and over the visible and near-infrared wavelength ranges. They found that, for some chars, the assumption of gray-body behavior was invalid. In spite of such findings, most ablation analyses utilize constant values for char surface emissivity and absorptivity. For Jovian entry, large amounts of ultraviolet radiation from the shock layer are predicted (see Fig. 20). In the case of air shock layers, much of the ultraviolet radiation is absorbed by ablation products. It now appears that such will not be the case for Jovian entry. It has been suggested that the

*Recent calculations by the same author, using new abosrption coefficient data for the polyatomic carbon species have indicated reductions in radiative heating of up to 80 percent. More work in this area is needed.

high energy, short wavelength ultraviolet radiation might penetrate deeply into the ablator rather than being absorbed at the surface as most ablation analyses currently assume. Wilson ans Spitzer's data show increasing absorptance as ultraviolet wavelengths are approached and seem to suggest high ultraviolet absorptance values but absorptance measurements in the ultraviolet are not presently available. If the radiation should penetrate into the char, present analyses could give grossly incorrect predictions of ablator response during Jovian entry. Finally, there is the question of mechanical char failure in H_2/He atmospheres. As mentioned previously, the most commonly observed type of char failure is intimately associated with oxidation reactions at the char surface. When tested in inert streams, even low-density chars resist mechanical failure up to high levels of pressure and shear. If the Jovian atmosphere is essentially 100 percent He, mechanical char failure may not be a big problem even at relatively high pressures and shears. If large amounts of H_2 are present, however, it is possible that H/C reactions at the char surface could cause a type of char failure similar to that caused by O/C reactions in air.

Finally, the extremely high heating rates associated with He atmospheres may force us to reorient our thinking about the relative importance of the various ablative energy accommodation mechanisms. For severe Earth entries and for Venusian entries, we found that char surface reradiation is likely to be the most important single mechanism. As pointed out previously, however, the amount of energy that can be reradiated is limited (by the sublimation of carbonaceous materials) to around 1200 W/cm^2. Hence, if the radiative heating rates reach levels of 12 000 W/cm^2, char surface reradiation will be relatively unimportant while sublimation will become relatively more important. It has been suggested that, for such extreme conditions, materials with high surface reflectivity (to reflect the incident radiation) rather than high emissivity might be desirable, if such materials can be found.

State of Technology

Present indications are that the percentage of vehicle gross weight required for the heat shield will be significantly greater for Jovian entry than for Earth, Venus, or Mars entries. Jovian heat-shield technology is in an early stage of development and many fundamental phenomena require study before the Jovian entry problem can be accurately assessed.

Simulation Ground and Flight Testing

Ground Testing

While many types of ground test facilities were used in the early days of ablation research, the great majority of present-day ablation tests are carried out in arc-heated wind tunnels. A typical arc tunnel, equipped to provide combined convective and radiative heating, is shown schematically in Figure 21. The test gas is passed through an electric arc where it is heated to high temperatures. It subsequently expands through a convergent-divergent nozzle and exits as a supersonic test stream in a free jet test section. The model to be tested is mounted on a swing arm which allows it to be inserted in the test stream after stable arc-jet operating conditions have been achieved and to be retracted from the stream after the desired test time has elapsed. The flow leaves the test section through a diffuser and flows through a heat exchanger to either a vacuum sphere or a steam ejector which provides the pressure ratio required to fill the supersonic nozzle. Arc heaters have been developed which operate well at high arc chamber temperatures (enthalpies) and low arc chamber pressures. Other heaters have been developed which operate at high chamber pressure but at modest temperatures. It has not been possible, however, to develop heaters which operate at both high temperatures and pressures, nor has it been possible to build a single heater that operates well over the entire range of possible conditions. Hence, most test complexes have several arc heaters, with each heater equipped with several nozzles of varying expansion ratio to provide a reasonably wide range of test conditions.

As shown in Figure 21, some test facilities are equipped with radiation sources (usually high-pressure arc lamps or lasers) so that the test models can be exposed to combined convective and radiative heating. The range of test conditions currently available is shown in Figure 22(89,90). This range of test conditions was determined for tests conducted in air and there is some question as to whether this extensive a range of test conditions is available in gases such as CO_2, H_2, and He. For many years there were persistent reports of electrode failures resulting from CO_2 operation, but most facilities now seem to be capable of routine operation on CO_2 and the test conditions obtained are generally comparable to those obtained with air. All arc tunnels seem to run very well on He but, because of the safety problems involved, few tests have been run to date in H_2 or H_2/He mixtures. Some selected values of available radiative flux capabilities are also shown on Figure 22 to give an indication of the combined heating capabilities now available. Also presented on Figure 22 are representative Mars, Venus, and Jupiter entry conditions(89). As can be seen from Figure 22, our present ground-test facilities can produce conditions typical of Martian entry but not those typical of Venusian entry. Needless to say, Jovian entry conditions are considerably beyond our present test capabilities.

Even within the range of available test conditions, however, ground facilities do not actually simulate reentry flight conditions. This lack of simulation can be illustrated by considering the stagnation-point heating relation

$$\dot{q}_c \; \alpha \; \left(h_s \frac{P_{t_2}}{\sqrt{R_{eff}}} \right) \; \alpha \; \left(V_\infty^2 \frac{P_{t_2}}{\sqrt{R_{eff}}} \right)$$

Ground-test models are usually much smaller than the flight vehicles they represent. Hence, if the heating rate is the same for both, then either the pressure or the velocity must differ. This illustrates one of the basic difficulties of ground ablation testing. One or two aspects of a given flight condition can be duplicated, but when they are, other aspects, often important ones, must be in error. Because of this lack of complete simulation, ground-test results cannot be used directly to predict in-flight ablative behavior. Instead, the ground tests are used to define basic ablation mechanisms and to verify a theoretical ablation model (usually a digital computer code). The theoretical ablation model is then used to predict in-flight behavior.

Another problem associated with the small size of ablation test models is that multidimensional ablation effects can become important in ground tests even though they are negligible for the flight conditions that the ground tests are supposed to represent. One example of such an effect is the reduction in transpiration cooling effect caused by the pyrolysis gases flowing laterally through the char to the low pressure regions around the shoulders of the model rather than flowing perpendicular to the char surface as is assumed in one-dimensional ablation analyses. This effect, which can result in increased ablation rates, is described in Reference (23). Another effect of this type arises when lateral temperature gradients (due to small model size) become large enough so that significant amounts of heat are conducted laterally away from a given point on the char surface rather than perpendicularly to the char surface into the decomposing ablator below.

In Figure 22 it was seen that present radiative heating facilities cannot produce fluxes as high as those expected during advanced (especially, Jovian) planetary entries. These facilities also have another shortcoming which is possibly more important than the insufficient flux levels. That is, the spectral distribution of radiation from the lamps or lasers differs greatly from that of the gases in a radiating shock layer. This is illustrated in Figure 23. From this figure it is seen that the facilities are particularly deficient in the short wavelength ultraviolet region which constitutes a significant part of the shock-layer radiation. As mentioned before, it has been often suggested that ablation materials may respond differently to ultraviolet radiation than to visible or infrared radiation. It is possible that the ultraviolet may penetrate deep into the ablator rather than being absorbed near the char surface (as is assumed in practically all existing ablation computer programs). Tests of models under ultraviolet radiation are needed to resolve this question. Another important phenomenon that cannot presently be studied in ground tests is the absorption of shock-layer radiation by ablation products. As was shown earlier, this energy absorption mechanism is quite significant for Earth entry and may be significant for many planetary entries (especially entries into predominantly CO_2 atmospheres). Since the incoming radiation is absorbed preferentially rather than uniformly across the spectrum (actually most of this absorption is in the ultraviolet), this phenomenon cannot be studied unless ground facilities with spectral distributions similar to those of an actual shock layer can be developed.

Flight Testing

Flight tests have played an invaluable role in the development of Earth-entry ablation technology. Flight tests are expensive and require large project teams and long lead times but, because of the previously mentioned difficulties in scaling ground-test ablation data to flight conditions, they are mandatory for cases (such as manned entry) where the ablator response must be known with great accuracy.

The great advantage of flight testing is the ability to expose the heat shield simultaneously to the correct levels of all the important entry parameters (i.e., heating rate, pressure, enthalpy, shear, etc.).

Since this situation cannot be achieved in ground testing, it usually happens that a given set of ground-test data can be satisfactorily explained by several theoretical ablation models, each of which predicts differing in-flight behavior for the ablator. One of the most significant contributions of a flight test is that it shows which (if any) of the proposed theoretical models is capable of predicting both ground and flight results.

The preceding comments apply when the flight test is carried out in the atmosphere of the planet for which the entry vehicle is designed. Suppose we are dealing with a vehicle designed for a Mars or Venus entry. Can meaningful flight tests of this vehicle be carried out in the Earth's atmosphere? This problem has been studied by several investigators[80,91,92]. As far as phenomena which are largely independent of atmospheric composition (i.e., decelerations, aerodynamic loads, etc.), the answer is yes. If one assumes a straight line ballistic trajectory(93) and an exponential atmosphere, it can be shown that, for the case where the entry velocities and ballistic coefficients are the same, we can select a path through the Earth's atmosphere along which the vehicle will encounter a density history equal to that which it would encounter along a different path in a different atmosphere. Beginning with this argument, Kennet(92) goes on to show that, along this equivalent trajectory, any quantity of the form

$$\rho_\infty^m V_\infty^n$$

can be duplicated so long as m and n do not depend on the atmospheric composition.

Laminar convective heating can be adequately correlated in terms of $\rho_\infty^m V_\infty^n$, but, of course, m and n will, in general, depend on the atmospheric composition. For air and CO_2, N_2 mixtures, however, the correlations are nearly identical ($\simeq$ 10 percent difference) and, hence, it appears that for Martian and Venusian entries, the laminar connective heating could be adequately simulated. There also appears to be a possibility of simulating transition and turbulent heating in CO_2-N_2 atmospheres, but this question has not been completely resolved at present.

Radiative heating, on the other hand, does not lend itself to simulation by these techniques. Since many of the most significant radiative heating phenomena (self-absorption, absorption by pyrolysis gases, possibly ablator response) are strongly dependent on the spectral distribution of the radiative flux, which is, in turn, strongly dependent on the chemical composition of the shock layer, this is to be expected. In Reference (80), Spiegel, Wolf, and Zeh carried out a detailed study of the degree to which the ablative response of a Venus-entry-vehicle heat shield could be studied and simulated in an Earth entry flight test. They found that convective heating could be simulated quite well, but that the radiative heating differed significantly, both with regard to flux level and spectral distribution, from that calculated for an actual Venus entry. Spiegel, Wolf, and Zeh went on to compute the in-depth response of a high-density phenolic-nylon heat shield for the Venusian and equivalent Earth entries and found that the ablator response was quite similar in spite of the differences in radiative heating. This would seem to imply that such an Earth entry might provide a proof test for a Venusian-entry heat shield. Such a conclusion, however, is predicated on the assumption that the Venusian-entry (and Earth-entry) radiative heating and the response of the ablator to it has been accurately described by the theoretical models used in the analysis. This is precisely the assumption that we wanted to check with a flight test in the first place.

Hence, it appears that the simulation of planetary-entry heat-shield response by an Earth-entry flight test is only feasible for missions involving only convective heating. An out-of-orbit Martian entry is an excellent example of such a mission. Because of the low heating rates involved and the fact that the most likely candidate materials (filled elastomers) are not greatly sensitive to shock layer composition at the resulting low surface temperatures, it appears that nearly complete simulation could be achieved. For higher velocity Mars-entry missions and for missions to Venus and the giant planets, however, Earth entry flight tests do not appear capable of simulating the significant radiative phenomena.

Concluding Summation

A review of ablative heat-shield technology for planetary entries has been carried out to assess the present state of the art and to identify areas in which further research is needed.

To identify important phenomena and to provide a basis for comparison, research on Earth-entry heat shields was summarized as a first step in the review. Emphasis was placed on entries characterized by large radiative and convective heating rates, since less severe entries can be considered as limiting cases (the case of negligible radiative heating, for instance) of this more complex problem.

It was shown that realistic assessments of heating and ablator response for this type of entry required a coupled analysis in which convective and radiative processes in the shock layer influence one another, and both the radiative and convective heating are significantly modified by the introduction of gaseous ablation products into the flow field. From such analyses it was shown that, for moderate-density charring ablators, the significant energy accommodation mechanisms are: pyrolysis gas enthalpy increase, char surface reradiation, and the reduction of convective and radiative heat flux by the injection of ablation products into the flow field. Significant uncertainties were shown to still exist with respect to the chemical state of the pyrolysis gases (which determines their enthalpy content) and the calculation of radiative heating rates with absorption by ablation products. It was pointed out that most present-day ablation analyses assume that radiation of all wavelengths is absorbed at the char surface, even though little is actually known about the absorptance of ablative chars at other than visible wavelengths. Shock-layer spectra were presented that showed large amounts of radiant energy at vacuum ultraviolet wavelengths. It was suggested that radiation at these wavelengths might penetrate deeply into the char and cause ablative behavior to be significantly different from that presently being predicted.

The rate at which the ablator is consumed in accommodating the incident heating is largely determined by the char recession rate which results from chemical reaction with boundary-layer gases, sublimation, and mechanical char failure. Significant uncertainties were shown to exist in the prediction of all three of these phenomena.

The prediction of convective heating rates was found to be on fairly firm ground so long as the flow remains laminar. The prediction of boundary-layer transition and turbulent heating rates still involve significant uncertainties, however, with transition Reynolds numbers being known only to within an order of magnitude. It was also pointed out that, because of experimental difficulties, the response of ablators to turbulent flow is a relatively unexplored area of research.

For entries into tenuous atmospheres such as those of Mars and Mercury, it was found that heat-shield design was well within the present state of the art. Because of the low entry velocities typical of out-of-orbit entries, radiative heating was found to be insignificant and convective heating rates were found to be low. It was pointed out that, for these entries, the most important ablation-material characteristics were low density (for high insulating efficiency) and the ability to survive prelaunch and transit environments such as sterilization, cold soak, and space vacuum.

In considering entries into the atmosphere of Venus, it was found that, because of the comparable entry velocities, the significant phenomena were essentially the same as those for planetary return Earth entry. Hence, most of the uncertainties discussed for severe Earth entry with combined radiative and convective heating apply as well to Venusian entries.

Since the Venusian atmosphere is primarily CO_2 (rather than air), the spectral distribution of shock-layer radiation is quite different from that for air and it was pointed out that most of the radiation from a Venusian shock layer was in a spectral range where it could be strongly absorbed by ablation products. Since very few coupled analyses (including absorption by ablation products) have been carried out for Venusian entries, it was suggested that this might be a fruitful field of research.

It was pointed out that the relatively high char surface pressures and aerodynamic shears which result from ballistic Venusian entries suggest significant rates of mechanical char failure. Considerable work is required in the area of char surface chemical reactions in CO_2 test streams, with particular emphasis on char failure by coupled chemical-aerodynamic mechanisms. Reference to Earth entry research showed that the mechanical char failure problem cannot be overcome by simply increasing the density of the ablator.

Because of the high Venusian-entry heating rates, elastomeric materials such as those proposed for Mars entries were considered unsuitable for Venusian entries. Because of their inability to withstand inter-planetary transit environments, many of the materials most often considered for severe Earth entries may also be unsuitable for Venus missions. Hence, it was suggested that a new class of materials that could withstand both the transit environments and the severe entries might have to be developed.

Jovian entry was considered as being typical of entries into the atmospheres of the giant planets (Jupiter, Saturn, Uranus, etc.). Because of the large mass of the planet (hence, high escape velocities), Jovian entry speeds and the resulting heating rates (especially radiative heating rates) were found to be potentially many times higher than those for Earth entry. The extent of the increase can not be evaluated because of uncertainties in the ablation products and their absorption coefficients, but it is almost certain that the radiative heating rates will be very large. It was pointed out that the behavior of materials under such high heating rates may be quite different than at lower heating rates. The possibility of catastrophic failure cannot be discounted. It was shown that, because the amount of energy reradiated from the char surface is limited by material sublimation temperatures, char surface reradiation could be a less important energy accommodation mechanism for Jovian entries than it is for Earth or Venusian entries. The possible significance of mechanical char failure was shown to depend to a significant extent on whether the Jovian atmosphere is mostly H_2 (which will react with carbonaceous materials) or He (which is inert). The state of the art for Jovian entry heat shields was found to be in an early stage of development.

The test capabilities of present-day ground-based facilities were reviewed and their inability to completely simulate entry conditions was pointed out. It was pointed out, however, that valuable partial simulation is available and, in particular, small test models can be subjected to convective heating rates up to the levels experienced in typical Venusian entries. It was also shown that radiative heating rates comparable to those of Venusian entry should be available soon. It was pointed out, however, that one important aspect of the radiative heating, the spectral distribution, is not reproduced in any of the present radiative facilities. Because of this, it is not presently possible to validate existing theories of radiative heating with absorption by ablation products. It was shown that Jovian entry conditions are beyond the capabilities of present ground facilities.

The ability of Earth-entry flight tests to simulate entries into other atmospheres was reviewed. It was found that those phenomena, which are independent of atmospheric chemical composition, could be adequately simulated but that phenomena which depend strongly on shock-layer composition (such as radiative heating and char surface phenomena) could not be simulated. Hence it appeared that nearly complete simulation of an out-of-orbit Mars entry was possible (since char surface reactions are not too important at the low temperatures involved), but that the value of Earth entry tests in studying Venusian and Jovian entry phenomena is limited.

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TABLE 1—Planetary atmospheres

	Mercury	Mars	Earth	Venus	Jupiter
Surface Pressure, atm	1×10^{-3}	5×10^{-3}	1	167	1×10^{5}
Surface Temperature, ^{o}K	200	130	288	770	2000
Tropopause Altitude, km	11	19	11	55	20 (above cloud top)
Stratosphere Temperature, ^{o}K	122	100	217	220	86
Density Scale Height, km[a]	6	5	7.73	6	18
Molecular Weight	40	44	29	42-44	3.4
Composition	Ar, CO_2	CO_2	O_2, N_2	CO_2, N_2	He, H_2
Entry Velocities, km/sec	3-4[b]	4-8[b]	8-20[c]	7-15[b]	40-60[b]

[a] Stratosphere
[b] Orbital to escape
[c] Orbital to planetary return

TABLE 2—Maximum heating rates, integrated heat loads, pressures, and shears for sphere-cone vehicles entering the Martian atmosphere

Trajectory				Stagnation Point					Cone Edge						Ref.
Entry Mode	$\frac{M}{C_D A}$	$-\gamma_e$	V_e	$\dot{q}_c$	$\dot{q}_R$	Q_c	Q_R	P	$\dot{q}_c$	$\dot{q}_R$	Q_c	Q_R	τ	P	
	$\frac{g}{cm^2}$	deg	$\frac{km}{sec}$	$\frac{W}{cm^2}$	$\frac{W}{cm^2}$	$\frac{J}{cm^2}$	$\frac{J}{cm^2}$	Atm	$\frac{W}{cm^2}$	$\frac{W}{cm^2}$	$\frac{J}{cm^2}$	$\frac{J}{cm^2}$	$\frac{N}{m^2} \times 10^{-2}$	Atm	
D	5.5	21	5.8	68	5.5	3270	329	0.16	31	2.1	1480	132	< 0.5	—	60
D	5.5	21	5.8	125	6.0	2560	176	.41	57	2.5	1160	70	< .5	—	60
O	4.7	15.7	4.6	26	N	2350	N	.05	12	N	1060	N	< .5	—	60
O	—	10.9	4.0	7.2	N	—	N	.03	2.5	N	—	N	—	0.02	61
O	—	14.1	4.6	13	N	—	N	.04	4.1	N	—	N	—	.03	61
O	—	20	4.6	27	N	—	N	.22	9.1	N	—	N	—	.17	61
O	—	14.1	4.6	17	N	—	N	.07	5.7	—	—	—	—	.05	61
O	3.1	15	4.6	18	N	—	N	.01	—	—	—	—	—	—	62
O	3.1	15	4.6	34	N	—	N	.03	—	—	—	—	—	—	62
MSSR	31.0	21	10.7	785	510	—	—	.32	—	—	—	—	—	—	62
D	3.1	90	7.0	176	18	1790	77	.39	—	—	—	—	—	—	63
D	3.1	38	7.0	136	9.1	2380	63	.22	34	43	—	—	—	—	63
D	3.1	20	7.0	91	2.3	3820	47	.10	23	43	—	—	—	—	63

D Direct entry
O Out of orbit entry
MSSR Mars surface sample return mission
N Negligible
— Data not available

Table 3 —Maximum heating rates, integrated heat loads, pressures, and shears for sphere-cone vehicles entering the atmosphere of Venus

	Trajectory			Stagnation Point					Cone Edge					Ref.
Entry Mode	$\frac{M}{C_D A}$	$-\gamma_e$	V_e	$\dot{q}_c$	$\dot{q}_R$	Q_c	Q_R	P	$\dot{q}_c$	$\dot{q}_R$	Q_c	Q_R	τ	
	$\frac{g}{cm^2}$	deg	$\frac{km}{sec}$	$\frac{W}{cm^2}$	$\frac{W}{cm^2}$	$\frac{J}{cm^2}$	$\frac{J}{cm^2}$	Atm	$\frac{W}{cm^2}$	$\frac{W}{cm^2}$	$\frac{J}{cm^2}$	$\frac{J}{cm^2}$	$\frac{N}{m^2} \times 10^{-2}$	
VMF	9.4	45	13.4	4500	15450	—	—	7	—	—	—	—	—	89
D	9.4	45	12.2	2800	4550	—	—	6	—	—	—	—	—	89
O	4.7	45	11.0	1480	398	—	—	3	—	—	—	—	—	89
O	9.4	45	11.1	2600	590	8180	1135	—	1820	1080	4100	2260	—	80
VMF	4.7	40	13.4	2700	4200	7500	6600	1.8	—	—	—	—	33.0	69
D	4.7	40	11.6	1650	740	5400	1700	1.3	1870	738	3980	1250	24.0	69
O	4.7	40	9.8	910	284	3520	910	1	—	—	—	—	19.1	69
D	5.4	35	11.6	1640	727	—	—	3	1820	660	—	—	24.0	60
O	5.4	30	9.8	885	258	—	—	1	955	193	—	—	16.3	60

VMF Venus-Mercury flyby
O Out of orbit
D Direct entry
— Data not available

Table 4 —Model atmospheres of Jupiter
(compositions are in weight percent)

Parameter	Kuiper Model (A)	Kuiper Model (B)	Öpik Model (O)	Spinrad and Trafton Model (K)	Sample Model (S)
H_2 %	78.25	38.78	2.3	60	44.83
He %	21.65	60.94	97.2	36	53.95
Ne %	0.07	—	0.39	3	0.22
CH_4 %	0.02	0.28	0.06	0.67	0.26
NH_3 %	—	—	—	0.33	0.08
Ar %	0.01	—	0.04	—	0.01
T* °K	165	168	156	150	160
P* atm	24	2	11	2.8	5.5
H* km	21.45	16.41	12.27	14.05	16.12

*At top of cloud layer (from ref. 56)

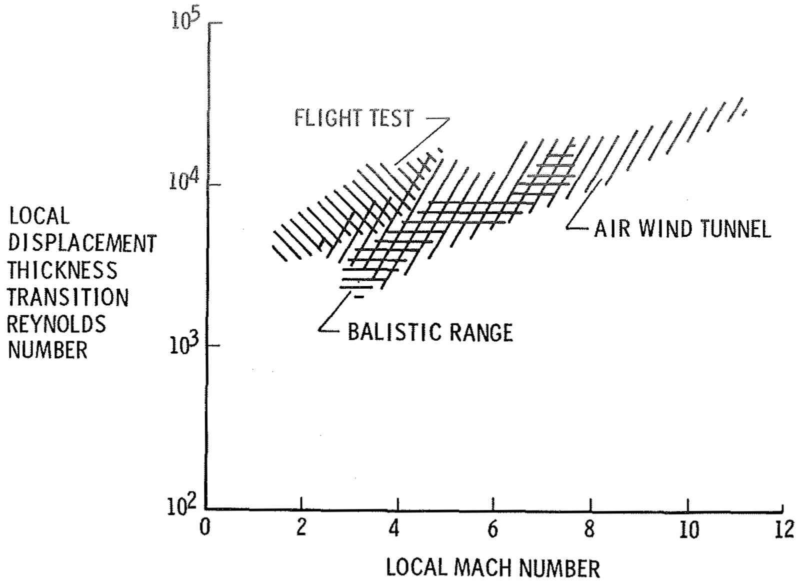

Figure 1.- Survey of transition data for nonablating sharp cones (from Ref. 11).

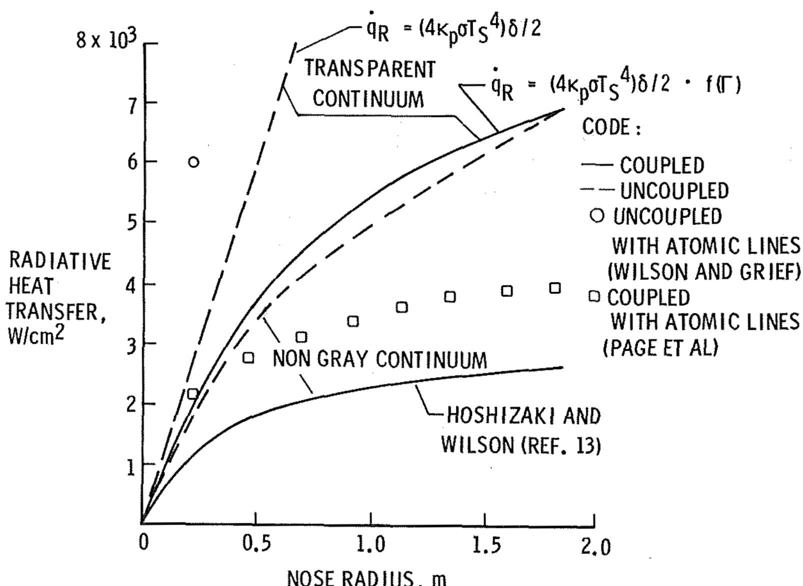

Figure 2.- Results for stagnation-point radiative heat transfer reflecting various stages of the state-of-the-art; V_∞ = 15.24 km/sec altitude = 61 km (from Ref. 12).

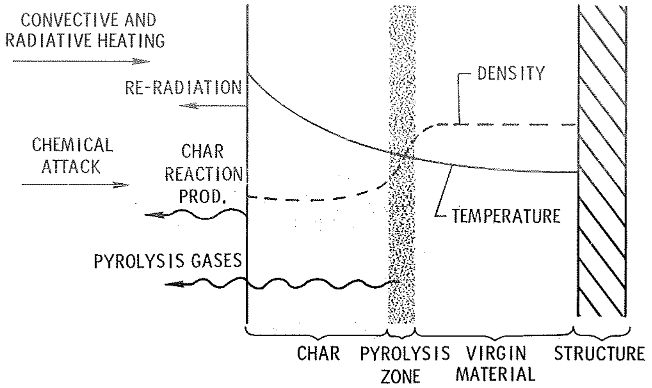

Figure 3.- Ablative mechanisms.

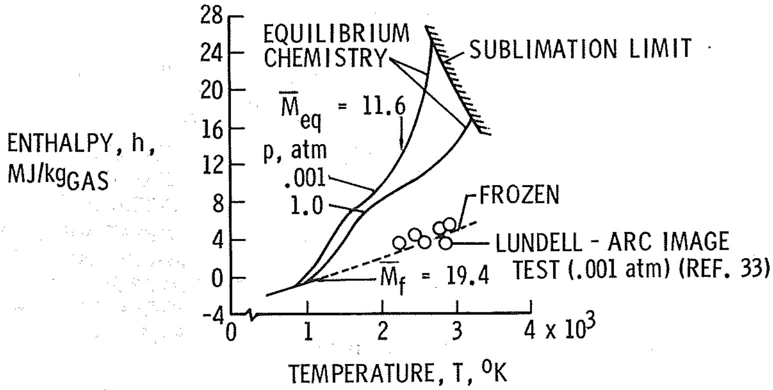

Figure 4.- Pyrolysis gas enthalpy as a function of chemical state (from Ref. 47).

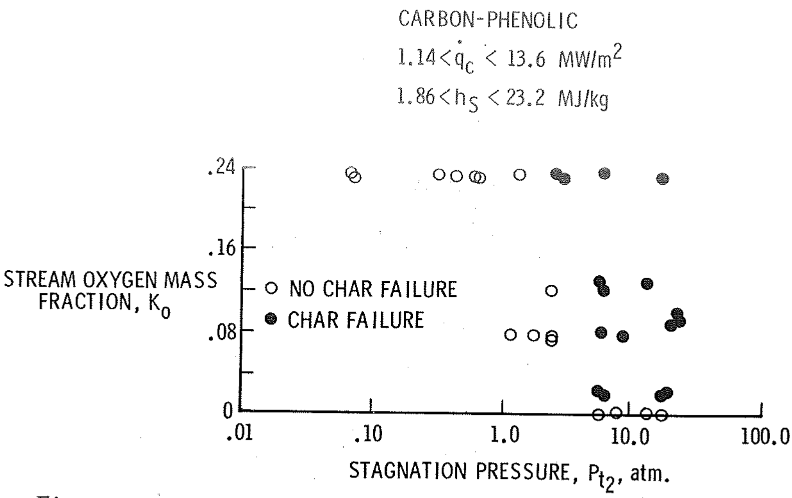

Figure 5.- Mechanical char failure for a carbon phenolic ablator (from Ref. 42).

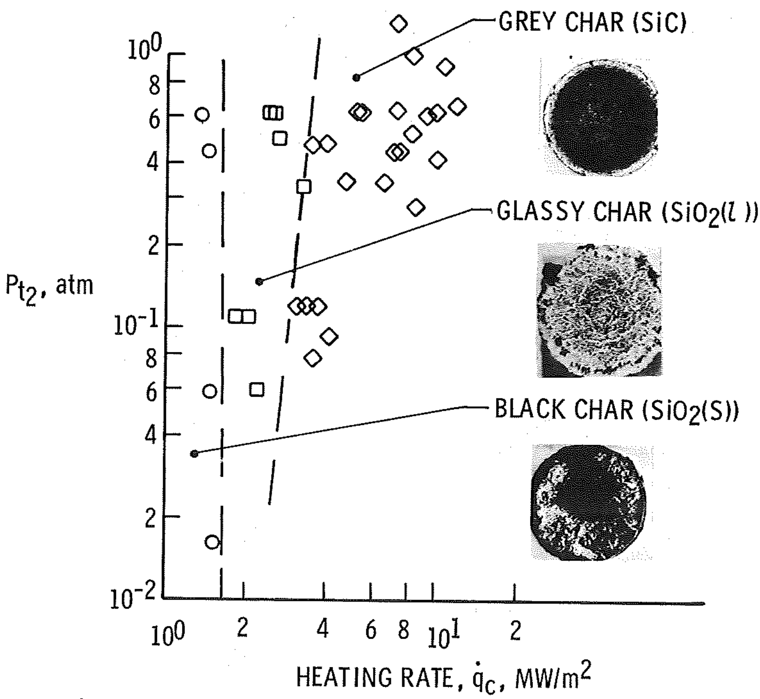

Figure 6.- Char surfaces produced by silicon dominated ablator (from Ref. 46).

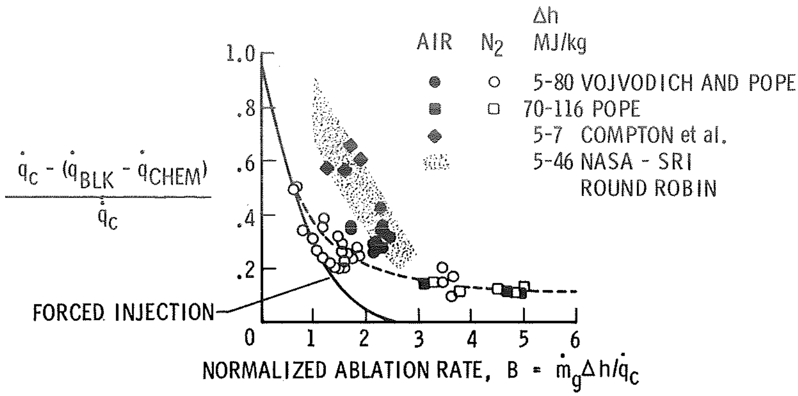

Figure 7.- Reduction of convective heating by pyrolysis gas injection (from Ref. 47).

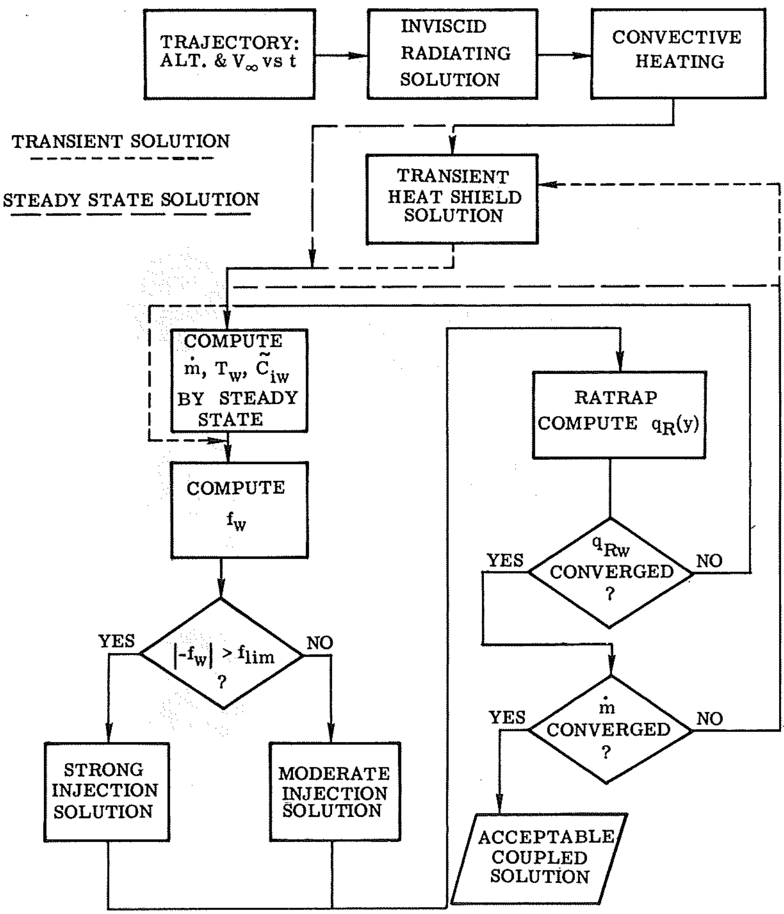

Figure 8.- Logic for coupled analysis (from Ref. 50).

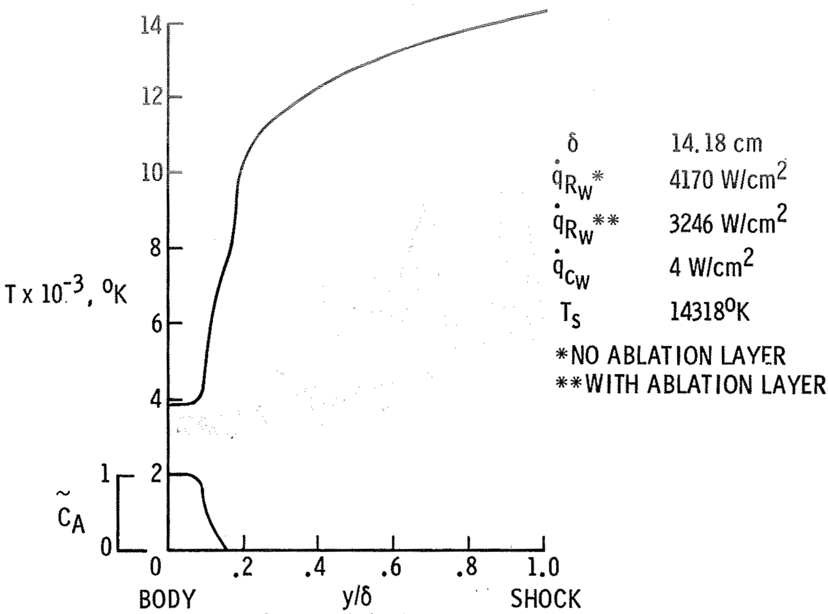

Figure 9.- Temperature and ablation products profile. Stagnation streamline of 304.8 cm radius sphere, V_∞ = 15.24 km/sec, 61 km altitude, carbon phenolic injection, $\frac{\rho_w V_w}{\rho_\infty V_\infty} = 0.2$, T_w = 3600° K (from Ref. 50).

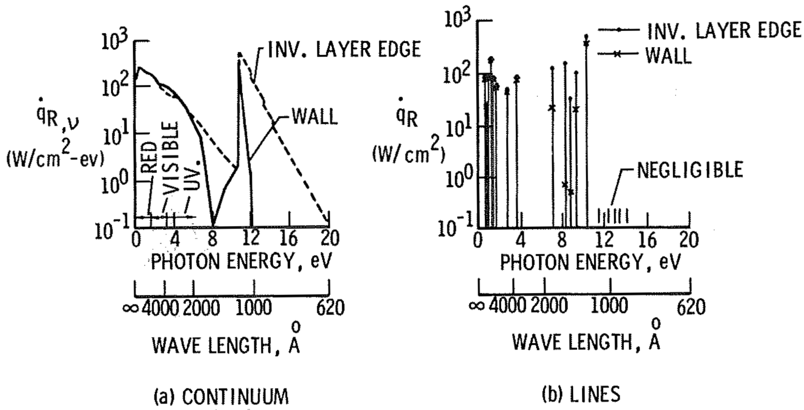

Figure 10.- Spectral distribution of radiation heat flux due to line and continuum processes (from Ref. 50).

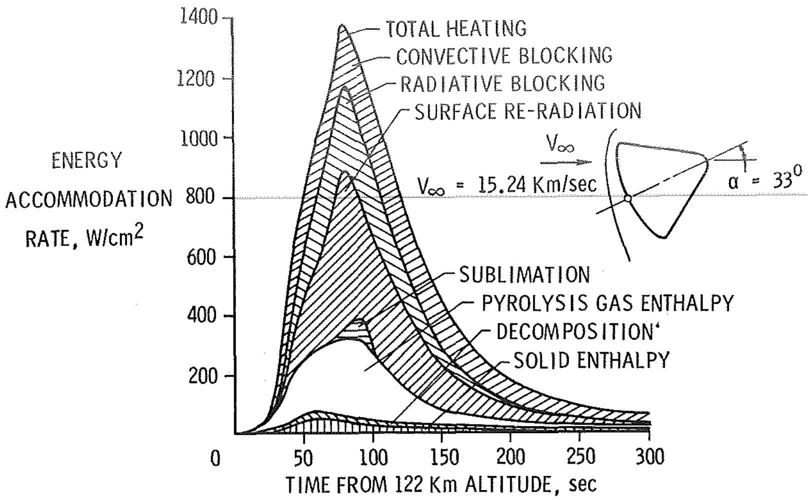

Figure 11.- Ablative energy accommodation mechanism for phenolic-nylon ablator on Apollo-like planetary-return Earth entry vehicle (from Ref. 49).

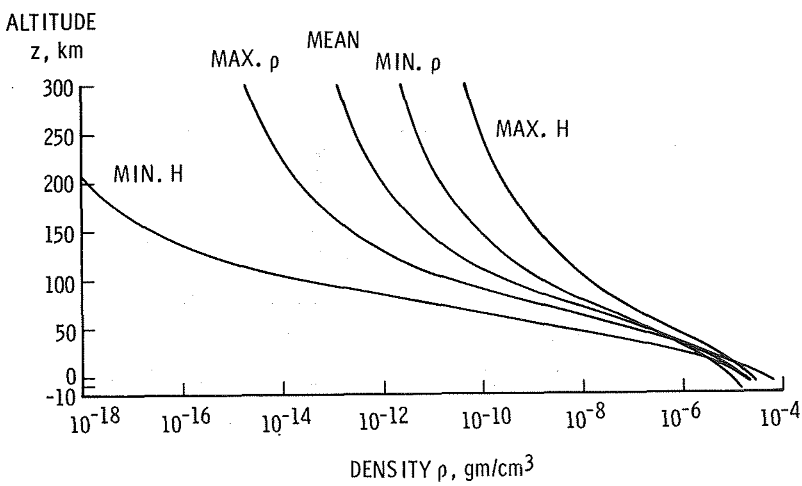

Figure 12.- Density profiles for five Martian atmospheric models.

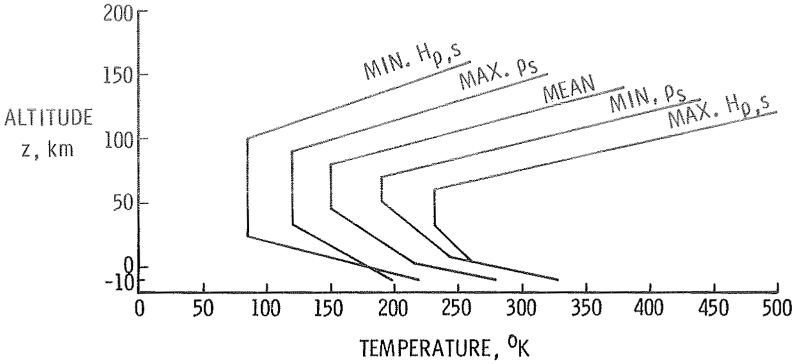

Figure 13.- Temperature profiles for five Martian atmospheric models.

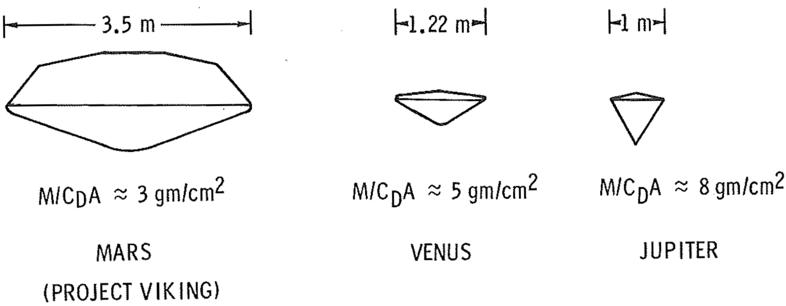

Figure 14.- Configurations for typical planetary entry vehicles.

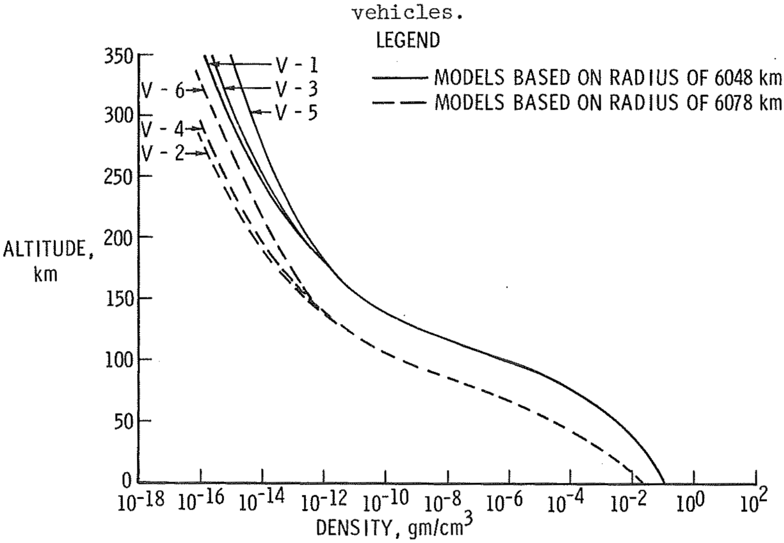

Figure 15.- Density versus altitude for models of Venus atmosphere (from Ref. 55).

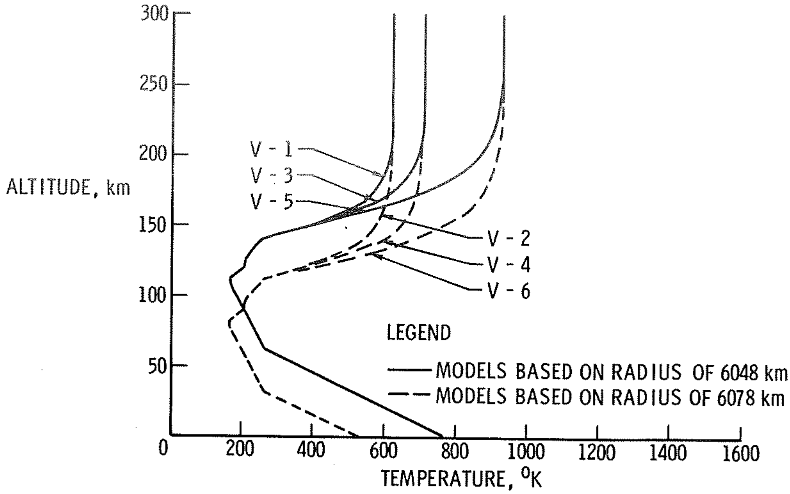

Figure 16.- Temperature versus altitude for models of Venus atmosphere (from Ref. 55).

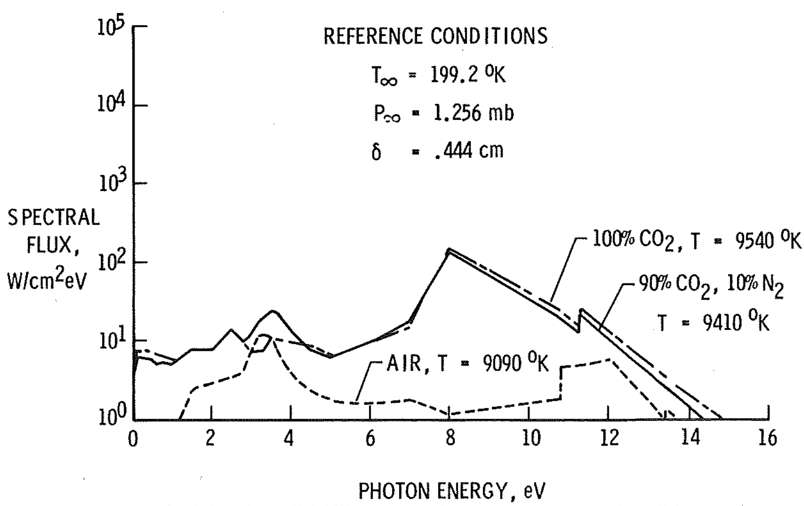

Figure 17.- Comparison of spectral flux from air and proposed Venusian atmospheres.

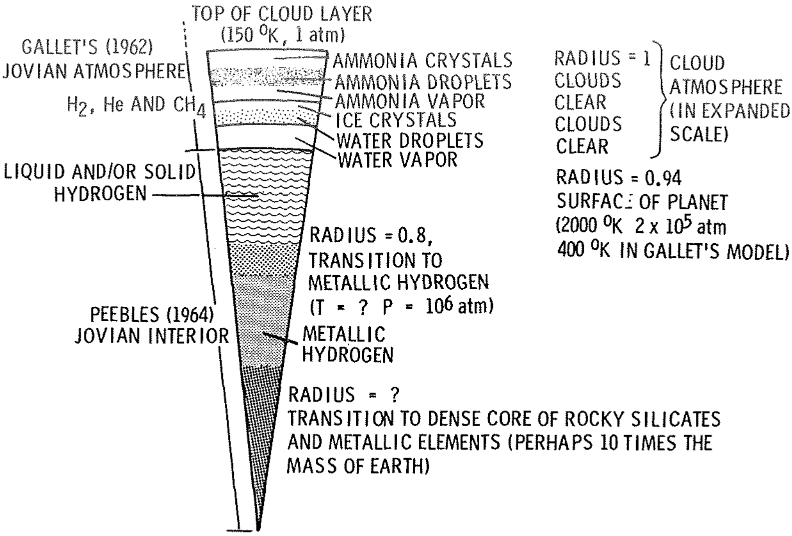

Figure 18.- Jovian cloud layers proposed by Gallet overlying Jovian interior by Peebles (from Ref. 57).

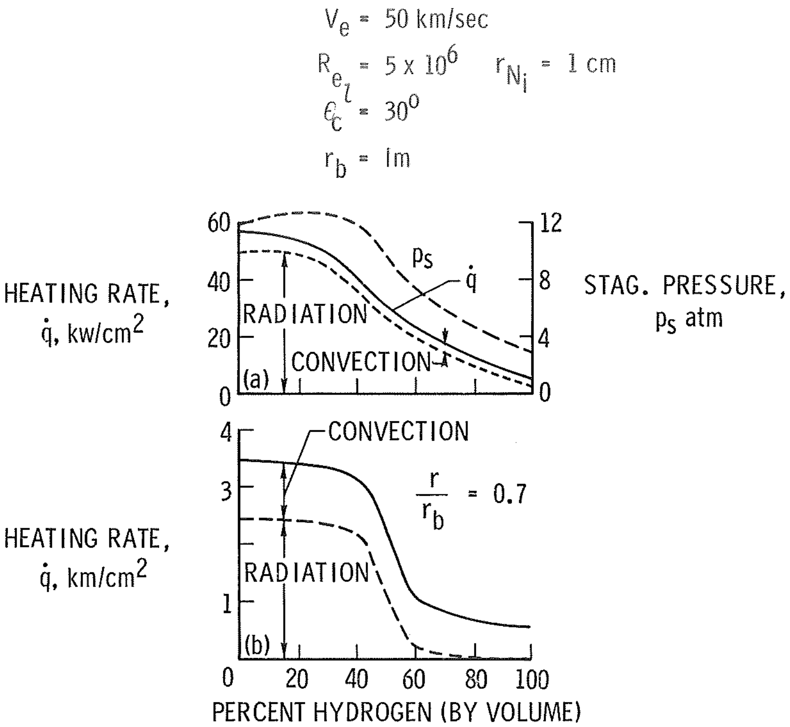

Figure 19.- Maximum heating rates and stagnation pressures for conical body during shallow ballistic entry; (a) stagnation point (ablating nose), (b) flank (at 0.7 of length point). (From Ref. 83.)

2

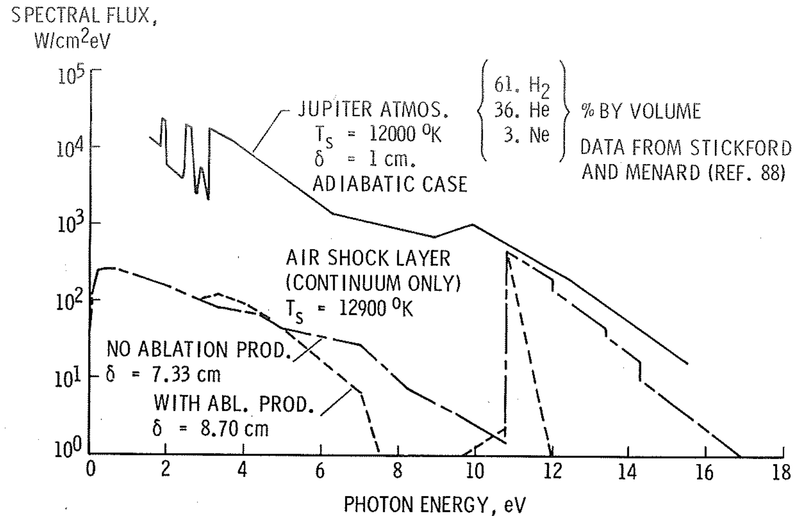

Figure 20.- Comparison of typical shock layer spectra for Jovian and Earth entries.

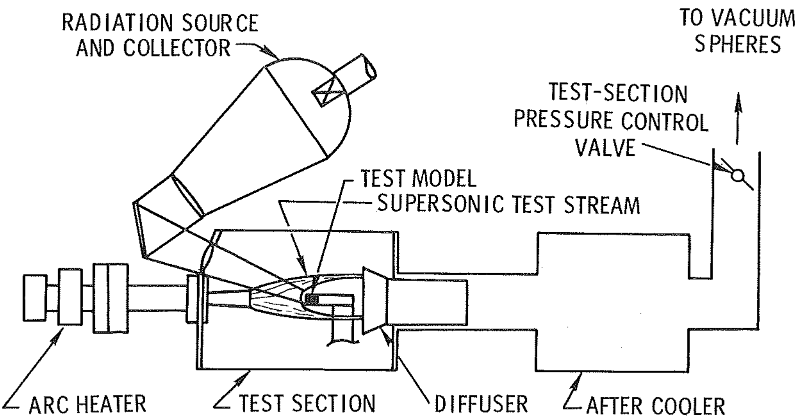

Figure 21.- Schematic of ground test facility.

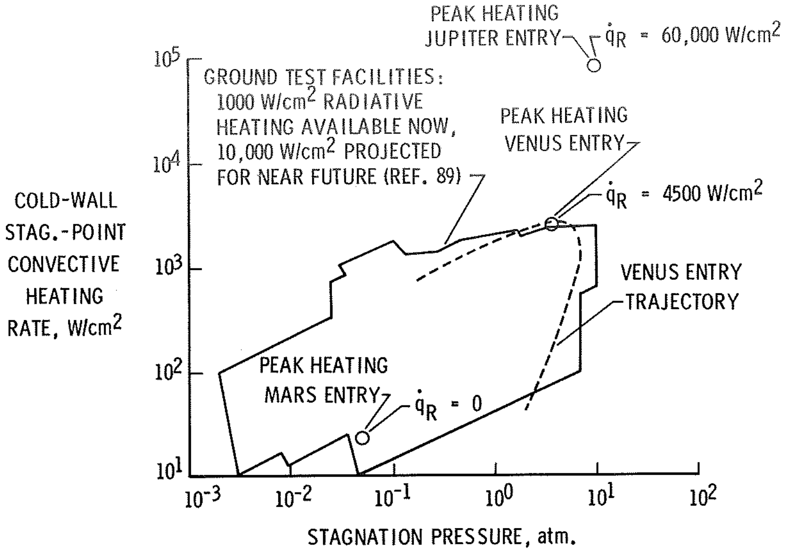

Figure 22.- Ground facility test capabilities.

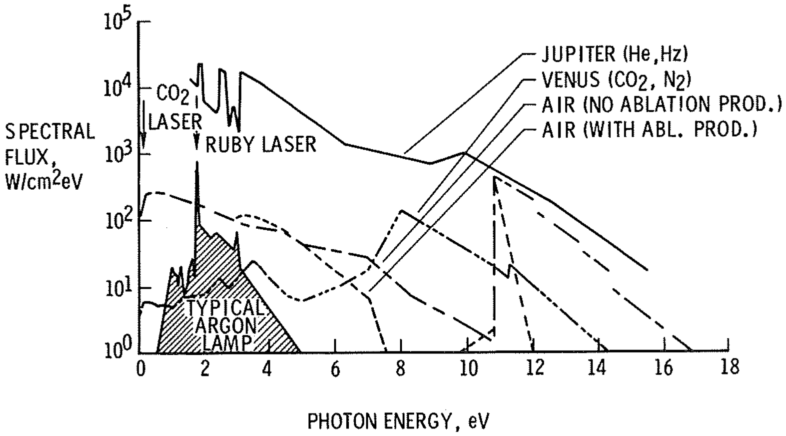

Figure 23.- Comparison of typical shock-layer spectra for Varius planetary atmospheres with spectra for ground test radiation sources.